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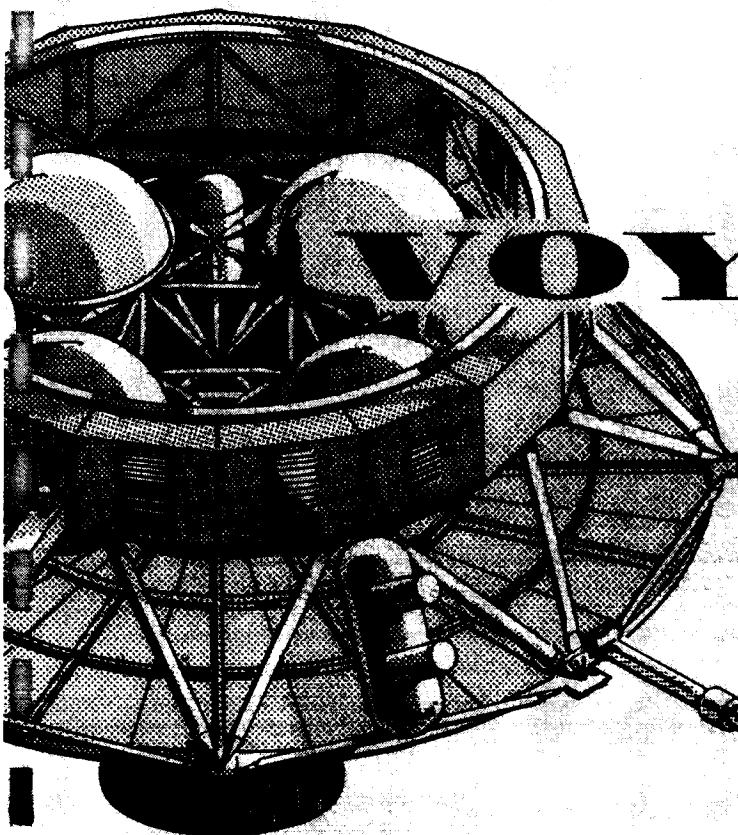
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VOYAGER

**spacecraft
system
studies**

PHASE B, TASK D



Prepared for

GEORGE C. MARSHALL SPACE FLIGHT CENTER
HUNTSVILLE, ALABAMA

THE **BOEING** COMPANY
SPACE DIVISION
SEATTLE, WASHINGTON

D2-115002-4

VOYAGER
SPACECRAFT SYSTEM STUDY

FINAL TECHNICAL REPORT
PHASE B, TASK D

VOLUME IV
SELECTED ENGINEERING TASKS

D2-115002-4
OCTOBER 1967

Prepared For :

NATIONAL AERONAUTICS AND SPACE ADMINISTRATION
GEORGE C. MARSHALL SPACE FLIGHT CENTER
HUNTSVILLE, ALABAMA

UNDER
CONTRACT NO. NAS8-22602

THE BOEING COMPANY ● SPACE DIVISION ● SEATTLE, WASHINGTON

FOREWORD

This series of documents summarizes the work performed under the George C. Marshall Space Flight Center contract, NAS 8-22602 entitled, "Voyager Spacecraft System, Phase B, Task D." The work was performed over the period June 16 through October 16, 1967.

The contracted work consisted of engineering studies leading to a definition of a Voyager Mars spacecraft system capable of performing the 1973 mission. To ensure flexibility of design, additional analyses were conducted to determine the adaptability of the 1973 spacecraft to perform the 1975-1977-1979 Mars missions. The 1973 flight spacecraft definition was used to identify the operational support equipment including mission-dependent equipment requirements and the software necessary to satisfactorily conduct the 1973 mission operations. Logistics considerations were identified for the 1973 system from point of manufacture through launch operations.

The contract also required the completion of five selected engineering tasks that were designed to highlight key areas and lead to specific conclusions and recommendations.

The detailed results of the contracted work is contained in the following reports:

- Summary Report Volume I
D2-115002-1
- Mission/System Requirements and Analyses Volume II
D2-115002-2
- Spacecraft Functional Description Volume III
D2-115002-3
- Selected Engineering Tasks Volume IV
D2-115002-4

CONTENTS

<u>Section</u>	<u>Page</u>
1.0 INTRODUCTION AND SUMMARY	1-1
1.1 Background and Objectives	1-1
1.2 Summary	1-2
1.2.1 Mission-Dependent Equipment (MDE) Definition	1-2
1.2.2 Voyager Program Test Flight	1-3
1.2.3 Science Payload Evolution	1-4
1.2.4 Particulate Contamination Considerations	1-5
1.2.5 Photoimaging Considerations	1-6
2.0 MISSION-DEPENDENT EQUIPMENT DEFINITION	2-1
2.1 Objectives	2-1
2.2 Scope	2-1
2.3 Approach	2-1
2.4 Results	2-2
2.4.1 Types and Quantities of Downlink Data	2-2
2.4.2 Types and Quantities of Uplink Data/Commands	2-13
2.4.3 Anticipated Real-Time Contingency Actions	2-13
2.4.4 Mission-Dependent Hardware	2-19
2.4.5 Mission-Dependent Software	2-22
2.5 Conclusions and Recommendations	2-28
3.0 VOYAGER PROGRAM TEST FLIGHT	3-1
3.1 Objectives	3-1
3.2 Scope	3-1
3.3 Approach	3-1
3.4 Results	3-2
3.4.1 Flight Test Requirements	3-2
3.4.2 Candidate Test Flight Profiles	3-5
3.4.3 Evaluation of Candidate Test Flights	3-6
3.5 Conclusions	3-15
4.0 SCIENCE PAYLOAD EVOLUTION	4-1
4.1 Objectives	4-1
4.2 Approach	4-1
4.3 Results	4-1
4.3.1 Science Objectives	4-1
4.3.2 Measurement Methods	4-5
4.3.3 Candidate Experiments	4-8
4.3.4 1973 Hypothetical Science Payload	4-8
4.3.5 Experiment Evolution	4-12
4.3.6 Experiment Characteristics	4-22
4.3.7 Science Payload Impact	4-44
4.3.8 Significant Changes to Spacecraft	4-50
4.3.9 Selected Science Payload Considerations	4-50
4.4 Conclusions and Recommendations	4-56

<u>Section</u>	<u>Page</u>
5.0 PARTICULATE CONTAMINATION CONSIDERATIONS	
5.1 Objectives	5-1
5.2 Scope	5-1
5.3 Approach	5-1
5.4 Results	5-3
5.4.1 Subsystem Requirements	5-3
5.4.2 Cleaning Processes and Clean Room Facilities	5-7
5.4.3 Cost Data	5-12
5.5 Conclusions	5-13
6.0 PHOTOIMAGING CONSIDERATIONS	6-1
6.1 Objectives	6-1
6.2 Approach	6-1
6.3 Results	6-2
6.3.1 Imaging System Requirements	6-2
6.3.2 Description of Candidate Imaging Systems	6-3
6.3.3 Imaging Systems Analyses	6-7
6.3.4 Comparison of Candidate Imaging Systems	6-27
6.4 Conclusions	6-41

1.0 INTRODUCTION AND SUMMARY

This document is the fourth volume of the Phase B, Task D Final Technical Report on Voyager spacecraft system studies performed by The Boeing Company. These studies were conducted for the George C. Marshall Space Flight Center under NASA Contract No. NAS8-22602.

The document contains the results of studies conducted on the following five selected engineering tasks:

- 1) Mission-dependent equipment definition
- 2) Voyager program test flight
- 3) Science payload evolution
- 4) Particulate contamination considerations
- 5) Photoimaging considerations.

1.1 BACKGROUND AND OBJECTIVES

Voyager spacecraft system definition phase studies have been conducted since early 1965. These studies have included concept refinements, assessment of preliminary requirements, and system analysis tradeoffs. These studies resulted in the definition and functional description of a spacecraft system capable of performing the 1971 mission.

During Task D of the definition phase, the flight spacecraft was revised to reflect (1) a change in the first Mars Voyager launch opportunity from 1971 to 1973, and (2) a need to develop a spacecraft system that can be adapted to subsequent Mars Voyager missions in 1975, 1977, and 1979 with minimum modifications. As a result, the functional description of the flight spacecraft hardware subsystems was revised (see Volume III, D2-115002-3).

Concurrent with this effort, five specific studies were conducted on key elements of the Voyager spacecraft system other than the flight spacecraft hardware subsystems. These system elements will (1) influence the fulfillment of mission objectives (science payload evolution, including photoimaging), (2) affect resource estimates (particulate contamination considerations), (3) influence mission success (mission-dependent equipment), and (4) affect confidence (Voyager program test flight).

The objectives of the first task, mission-dependent equipment (MDE) definition, were to (1) establish the requirements imposed by the Mars Voyager 1973 spacecraft on MDE, and (2) identify the hardware and software subsystems that would satisfy the established requirements. Early definition of MDE is important, since Lunar Orbiter experience indicates that MDE software costs constitute a significant fraction of total program costs, and MDE is critical to the successful solution of unforeseen contingencies during mission operations.

The objective of the second task, Voyager program test flight, was to determine the benefits of such a flight using the Saturn IB launch vehicle. Such a test flight would increase the confidence in the ability of the spacecraft to perform the Voyager mission. This is because ground testing cannot simulate the full mission profile for a spacecraft the size and complexity of Voyager. However, such a test flight would be costly. Consequently, the level of assurance provided by a test flight must be carefully weighed against realistic costs.

The objectives of the third task, science payload evolution, were to (1) determine the evolution of the science payload from the 1973 mission to the 1975, 1977 and 1979 missions, (2) develop the physical and functional characteristics of the science payload experiments for each launch opportunity, and (3) determine the impact of the payload evolution on the spacecraft. This study is necessary to ensure that the evolving experiment payloads satisfy the basic objectives of space exploration. An early identification of impact of the experiment payload on the spacecraft also is important. Experience gained from the Mariner '69 program indicates the difficulty of integrating even a few experiments into a spacecraft that was not designed specifically for those experiments.

The objectives of the fourth task, particulate contamination considerations, were to (1) determine the most reasonable level of particulate (nonbiological) contamination for the flight spacecraft and its subsystems, and (2) establish the impact of the required contamination control level on facilities and cost.

This study is important as evident most recently from the nearly disastrous effects of particulate contamination in the helium pressurization system on the Surveyor V mission. Also, for a payload the size of Voyager, clean room facility costs would be significant. Hence, the required level of particulate contamination should be determined with care.

The objectives of the fifth and last task, photoimaging considerations, were to (1) investigate and compare electrostatic tape, film, and vidicon photoimaging systems, (2) determine the impact of the photoimaging systems on the flight spacecraft, and (3) establish the highest achievable resolution of the photoimaging systems as a function of weight. The results obtained by Lunar Orbiter indicate that much can be learned about nature and evolution of extraterrestrial bodies by both high and medium resolution photography. The Voyager mission, however, is of longer duration than Lunar Orbiter and is constrained to higher orbital altitudes. The applicability of Lunar Orbiter photoimaging equipment to Voyager must therefore be examined carefully. Also, the high orbital altitudes and the planetary communication distances will result in significant impact on the spacecraft (e.g., weight) for the required surface resolution and coverage. This impact, therefore, must be established.

Details of the approach of studies performed for the above five engineering tasks are given in Sections 2.0 through 6.0. Key conclusions and recommendations resulting from the studies are included in the summary below.

1.2 SUMMARY

1.2.1 Mission-Dependent Equipment (MDE) Definition

MDE requirements were established by considering (1) types and quantities of

downlink data, (2) types and quantities of uplink data/commands, and (3) real-time contingency action resulting from both spacecraft and MDE malfunctions. From these it was established that the principal MDE hardware is required at the Deep Space Instrumentation Facility (DSIF) for demodulating and decommutating the telemetry stream. The upper subcarrier, which carries the high rate (orbital) science data, requires the following MDE hardware elements: filter, demodulator, synchronizer, digital-to-analog converter, block decoder, and a recorder. The lower subcarrier, which carries frame sync, engineering, and low rate (cruise) science data, requires the following additional MDE hardware elements: filter, demodulator, synchronizer, and a buffer-and-formatter. A test patch panel for the MDE hardware also is required.

MDE software development is required in the following five major areas: (1) telemetry and command data handling (TCD), (2) flight path analysis and command (FPAC), (3) spacecraft performance analysis and command (SPAC), (4) mission integration and control (MIC), and (5) simulation for software and system checkout and operational training. Most of these software programs are required at the Space Flight Operations Facility (SFOF). More than half of the required software programs could be obtained by modifying existing mission-independent software.

The MDE definition study led to the following conclusions:

- 1) Basic MDE requirements for the Voyager mission are similar to those for previous missions which used the Deep Space Net (DSN).
- 2) Up- and downlink Mars Voyager 1973 data rates will not result in unusual MDE requirements.

Some features of the Voyager MDE requirements are unique because of the following: (1) first Mars orbiting operations, and (2) simultaneous operation of two spacecraft. Therefore, the following specific MDE recommendations are made:

- 1) Spacecraft equipment simulation software should be developed that will (a) permit prediction of spacecraft status, and (b) furnish a "test bed" upon which mission operations personnel can evaluate and check out commands.
- 2) There should be separate display and command consoles in the SFOF for each spacecraft.
- 3) There should be a display which facilitates comparison between computed current spacecraft status and current telemetered spacecraft status.

1.2.2 Voyager Program Test Flight

Requirements for a Voyager program test flight were developed by (1) identifying critical 1973 mission events, (2) determining deficiencies in ground testing that could be fulfilled by a flight test, and (3) establishing where a flight test could reduce ground testing. Twelve flight test requirements were determined, including requirements for such tests as (1) propulsion interaction, (2) payload separation and sequencing, and (3) zero-g deployment. Three candidate test flights were considered which satisfy the identified flight test requirements, and are within the capability of the Saturn IB launch vehicle. The first test flight called for launching a planetary vehicle (PV) into an elliptical Earth orbit. Following capsule separation in orbit, the spacecraft propulsion module is used to inject the spacecraft into an escape trajectory towards the orbit of Mars, i.e., a deep space flight. The second test flight was similar to the first, except that following

capsule separation, the spacecraft was placed in a synchronous Earth orbit. The third test flight called for launching a planetary vehicle consisting of a flight spacecraft and a dummy capsule into an elliptical Earth orbit.

An evaluation of the three candidate test flights led to the following study conclusions:

- 1) A test flight increases the level of confidence for the Voyager program.
- 2) Reductions in ground testing and facilities attendant to a test flight are not significant.
- 3) The most valuable test flight is a deep space flight.
- 4) A test flight can support the 1973 Voyager mission.
- 5) Rescheduling is required to make the test flight compatible with the Voyager mission.
- 6) A Voyager program test flight increases program cost by approximately 50 to 100 million dollars.

1.2.3 Science Payload Evolution

The scientific objectives of Mars exploration--the origin and evolution of Mars and the solar system, and the origin and evolution of life--were examined. For these two basic objectives, the following six interrelated areas of inquiry were examined: (1) composition, (2) history, (3) exobiology, (4) differentiation, (5) activity, and (6) atmospheric dynamics. These areas of inquiry require experiments to determine chemistry, structure, processes, and vestiges (traces) for the atmosphere, crust, interior, and biosphere of Mars. Eighteen such experiments were identified. These experiments use to advantage the inherent capability of the spacecraft to perform remote measurements of radiation fields either emanated or reflected from the planet. Means for utilizing the growth capability of the experiments (e.g., increased resolution and coverage) were examined. From the above, the evolution of the science payload from 1973 to 1979 was determined. For each payload experiment, the following were then developed: (1) experiment objectives, (2) physical and functional characteristics, (3) instrument design characteristics, and (4) experiment requirements. From these, the impact of the weight, size, power, and data generation of the science payload on the flight spacecraft were established. Significant changes to the spacecraft resulting from the science payload evolution were identified.

The following conclusions and recommendation resulted from this study:

- 1) The photoimaging experiment is the most important single experiment contributing to the accomplishment of the basic space exploration science objectives.
- 2) The spacecraft best lends itself to remote electromagnetic radiation sensing
 - (a) over broad areas, (b) to high resolution, (c) through seasonal variations.

- 3) The subsatellite experiment (orbital experiment capsule) provides a means of measuring the Mars magnetic fields to as low as 0.25 gamma.
- 4) The evolution of the total science payload will increase its approximate weight from 400 pounds in 1973 to 600 pounds in 1975 and 1000 pounds in 1977 and 1979.
- 5) Evolution of the photoimaging experiment causes the greatest impact to the spacecraft. Its growth from 150 pounds in 1973 to over 600 pounds in 1977 and 1979 is justified by the need for higher resolution coverage of the planet's surface.
- 6) The future direction of experiment payload evolution depends on the results of early planetary measurements. If data returned from early missions indicate the possibility of life, then experiment emphasis will shift from orbiter to lander.
- 7) Improvements are required in the area of telecommunications to handle the much larger amounts of scientific data that will be generated in subsequent missions.
- 8) The centralization of data automation equipment (DAE) functions in the computer and sequencer (C&S) and data storage subsystem result in the following advantages; 1) equipment savings; 2) interface flexibility and simplicity; 3) increased command capability and flexibility; and 4) improved data buffering control. This centralization would be accomplished by 1) combining the DAE command functions of timing, scan platform control, instrument sequencing and power switching within the C&S; and 2) combining the DAE functions associated with instrument output data multiplexing, formatting, and buffering with similar functions in the telecommunication subsystem (data storage).

1.2.4 Particulate Contamination Considerations

Contamination control requirements were established for the Voyager spacecraft and its hardware subsystems. These included the following: (1) maximum size, number and type of allowable contamination, (2) critical hardware components, (3) contamination impact, and (4) particulate contamination control for various levels of testing. The following data were then established for each critical subsystem: (1) allowable contamination limit for each type of operation (e.g., fabrication, assembly, and test), (2) proposed techniques and equipment required to meet allowable contamination limits, and (3) type of facility required. Cost data were developed for clean room facilities, and, using Lunar Orbiter data, clean room operations. This study resulted in the following conclusions:

- 1) Clean room facilities are required for Voyager.
- 2) The most reasonable level of Voyager particulate contamination control requires a Class 100,000 clean room. Limited operations on a Class 100 bench within a Class 100,000 room also are required.
- 3) A wall-to-wall laminar flow type of clean room is preferred for Voyager because of its low cost (\$100/square foot for a 60-foot ceiling).
- 4) Clean room operations will increase fabrication, assembly, and test costs by approximately 14%.

- 5) Clean room and cleaning procedures, and training, are necessary to ensure the required spacecraft level cleanliness.
- 6) Existing (e.g., Lunar Orbiter) cleaning and clean room procedures can be adapted to Voyager.

1.2.5 Photoimaging Considerations

The high and medium resolution coverage requirements for photoimaging were established. The gross physical characteristics of film, vidicon, and electrostatic tape (ESTC) imaging systems capable of satisfying the requirements were defined. Nomographs were developed for key optical and imaging parameters to facilitate imaging system analysis. The three candidate imaging systems--film cameras, vidicons, and ESTC--were analyzed, including considerations of resolution, format, stereo, coverage, focal length, aperture, exposure time, optics design, weight, and smear and image motion compensation (IMC). The three candidate imaging systems were then compared on the basis of contribution to mission success and performance of mission objectives, including the impact of photoimaging resolution on spacecraft physical and functional characteristics. The highest resolution imaging capability for the three candidate imaging systems also was established.

This study resulted in the following conclusions and recommendations:

- 1) High resolution (1 to 10 meters) photoimaging coverage of a small fraction of Mars' surface (0.1 to 1%) and medium resolution (150 to 300 meters) photoimaging coverage of most of the planet's surface will satisfy the presently understood scientific objectives.
- 2) The film camera, vidicon, and electrostatic tape camera systems can satisfy the nominal photoimaging resolution and coverage requirements.
- 3) The film camera system provides the highest resolution of the three candidate imaging systems.
- 4) For the lowest allowable orbital altitude of 500 km (from planetary quarantine considerations), the highest resolution of a film camera is estimated as 0.5 meters. This resolution is limited only by the postulated scattering, turbulence, and aerosol phenomena of the Martian atmosphere and not by equipment capability. This 0.5-meter resolution is achievable with a film camera weighing less than 700 pounds.
- 5) The flight spacecraft weight will increase by 1800 pounds to accommodate the 700-pound film camera. This would allow for approximately 0.01% coverage of the planet at the 0.5-meter resolution over a 180-day orbital mission.
- 6) For the lowest allowable orbital altitude of 500 km, the highest achievable resolution of either the vidicon system or electrostatic tape system is estimated to be 5 meters due to sensor/lens limitations.
- 7) For the lowest allowable orbital altitude of 500 km, the film camera system will have the least weight for resolutions below 10 meters.

- 8) The vidicon imaging system will satisfy the medium resolution requirements for the least weight.
- 9) For high resolution imagery, modified Cassegrainian-type reflective telescopes, using folded optics, will provide the required effective focal length of 6 to 12 meters within the spacecraft envelope constraint.
- 10) Film processes have been developed with life capability in excess of 2 years under simulated space conditions.
- 11) The SEC vidicon, because of its high sensitivity, will result in the lightest optical system by comparison with other sensors.
- 12) Advanced 3000-line slow-scan vidicons (30 seconds) and high read-in-rate tape recorders (400,000 bps) should be developed to satisfy the requirements of the hypothetical photoimaging payload currently proposed for the 1973 mission.

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2.0 MISSION-DEPENDENT EQUIPMENT DEFINITION

2.1 OBJECTIVES

The primary objective of this task was to establish the requirements imposed by the Voyager Mars 1973 post-injection mission operations on the mission-dependent equipment (MDE). Another objective was a preliminary identification of the hardware and software that satisfy those MDE requirements.

2.2 SCOPE

Mission-dependent equipment is one of three major elements of the mission operation system (MOS). The other two major MOS elements are:

- 1) Existing equipment at the Deep Space Network (DSN), Air Force Eastern Test Range (AFETR), and Kennedy Space Center (KSC) assigned to support the conduct of Voyager mission operations.
- 2) Operations teams at the Space Flight Operations Facility (SFOF) which conduct mission operations from injection to the end of Mars orbital operations.

The Deep Space Network (DSN) consists of equipment and facilities required to support the tracking, communications, and data handling functions associated with deep space exploration. The net consists of (1) deep space instrumentation facilities (DSIF) at various locations around the world, (2) the space flight operations facility (SFOF) in Pasadena, California, and (3) a ground communications system (GCS) connecting these facilities.

Different missions and spacecraft require additions to the DSN. These additions ensure the compatibility of the DSN with any given mission. Additions can be either hardware (e.g., equipment required to decommutate and demodulate spacecraft telemetry) or software (e.g., computer programs for analyzing the spacecraft's flight path). Such hardware and software equipment additions to the DSN constitute the mission-dependent equipment.

2.3 APPROACH

To accomplish the primary task objective, the following data were developed as a function of mission time:

- 1) Approximate types and quantities of downlink data transmitted to the ground.
- 2) That portion of the downlink data required for real-time assessment of spacecraft status and performance.
- 3) Anticipated real-time contingency actions.
- 4) Quantity and types of uplink data/commands under both routine and contingency conditions.

Using the above data, requirements for MDE hardware and software were identified.

2.4 RESULTS

2.4.1 Types and Quantities of Downlink Data

The types and quantities of downlink data influence MDE design. The downlink data are received from the spacecraft telemetry subsystem.

The spacecraft telemetry subsystem provides acquisition, limited conditioning, and formatting of engineering and science data. The data are then modulated, mixed with the appropriate subcarrier, and transmitted to Earth by the radio subsystem. Two subcarriers, termed upper and lower subcarriers, are used for transmitting the data.

The telemetry subsystem acquires and processes data from each major spacecraft subsystem independently of other sources. It varies the contents of the data frame to accommodate the data requirements of a particular mission phase.

The types and quantities of data transmitted from the 1973 baseline spacecraft to the DSN are shown in Figure 2-1. The data are shown for each of the six telemetry modes selected for the baseline 1973 mission. Both lower and upper subcarrier data are included. The lower subcarrier contains five data types as described below:

- 1) Frame Sync -- which contains the sync, identification word and frame count.
- 2) Spacecraft Engineering -- which provides spacecraft status and performance information for the real-time assessments on the ground. (The information content of the spacecraft engineering data category is given in Table 2-1.)
- 3) Capsule Engineering -- which provides capsule status and performance information prior to capsule separation. The spacecraft sequences and receives capsule data trains for insertion into its telemetry stream.
- 4) Cruise Science -- which originates in instruments designed primarily for measuring the interplanetary environment. These instruments include: plasma probe, cosmic ray telescope, cosmic dust detector, radiation detector, and an ion chamber. These instruments also provide data during orbital operations.
- 5) Tape Recorder Playback -- consisting of (a) data stored during maneuvers and Earth occultation periods, and (b) solar flare data. The latter are automatically recorded and transmitted whenever on-board instrumentation senses that a predetermined level of solar activity has been reached.

The data rates for each of the above five data types are specified for each telemetry mode in Figure 2-1. From these rates, the total quantity of data for each data type can be determined as a function of mission time.

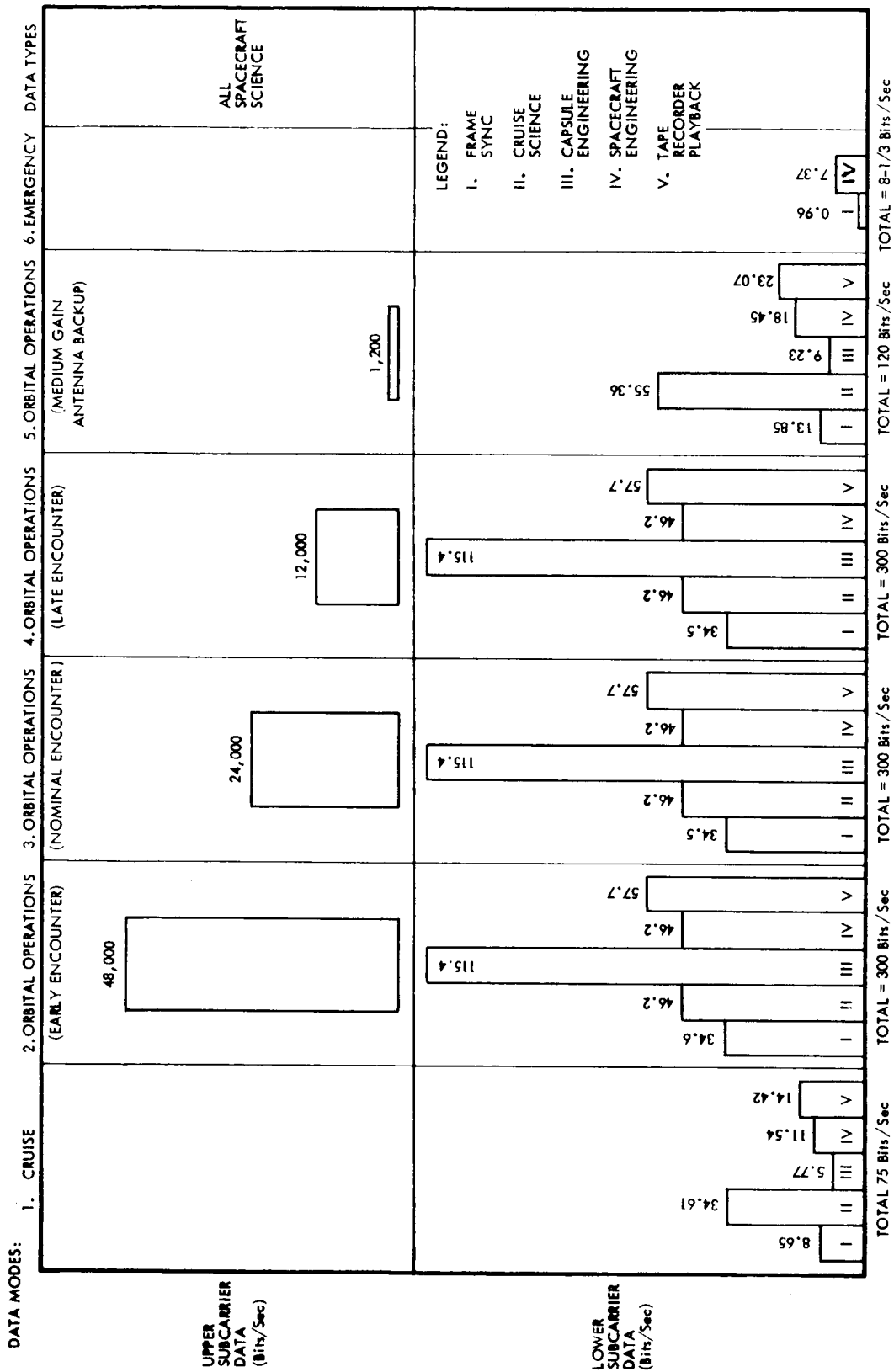


Figure 2-1: TYPES AND RATES OF SPACECRAFT - GENERATED DOWNLINK DATA

Table 2-1: SPACECRAFT ENGINEERING MEASUREMENT LIST
(Sheet 1 of 9)

MEASUREMENTS			SIGNAL RANGE
SOURCE	QTY.	UNITS	
ANTENNA			
High-Gain Antenna Deployment	1	position	1 bit
High-Gain Antenna Position	2	angle	0-5v
Low-Gain Antenna Deployment	2	position	1 bit
Med.-Gain Antenna Position	1	angle	0-5v
Capsule Relay Antenna Deployment	1	position	1 bit
Med. Gain Antenna Deployment	1	position	1 bit
STRUCTURAL AND MECHANICAL SUBSYSTEM			
Science Scan Platform Deployment	5	position	1 bit
Science Scan Platform Position	3	angle	0-5v
UV Platform Position	2	angle	0-5v
Capsule Separation	1	position	1 bit
Accelerometer	6	ft/sec ²	0-5v
Shroud Void Pressure	3	psia	0-5v
PV Separation	1	position	1 bit
Shroud Void Temperature	3	°F	0-5v
TEMPERATURE CONTROL SUBSYSTEM			
Louver Position	21	angle	0-5v
Radiator Plate Temperature	21	°F	0-5v
Solar Shield Temperature	2	°F	0-5v
Engine Heat Shield Temperature	2	°F	0-5v
Solar Panel Temperature	4	°F	0-5v
Spacecraft Skin Temperature	2	°F	0-5v
Antenna Temperature	3	°F	0-5v
Propulsion Subsystem Temperature	4	°F	0-5v
Science Scan Platform Temperature	2	°F	0-5v
PYROTECHNICS SUBSYSTEM			
Pyrotechnic Safe Arm	2	condition(cond)	1 bit
Pyrotechnic Event Signals	4	event	4 bit
Pyrotechnic Power Status	2	cond	2 bit

Table 2-1: SPACECRAFT ENGINEERING MEASUREMENT LIST
(Sheet 2 of 9)

MEASUREMENT			SIGNAL RANGE
SOURCE	QTY.	UNITS	
SCIENCE SUBSYSTEM			
Plasma Probe Temperature	2	°F	0-5 v
Cosmic Ray Telescope Temperature	2	°F	0-5 v
Cosmic Dust Detector Temperature	2	°F	0-5 v
Trapped Radiation Detector Temperature	2	°F	0-5 v
Ion Chamber Temperature	2	°F	0-5 v
Ultraviolet Spectrometer Temperature	2	°F	0-5 v
High Resolution Infrared Spectrometer Temperature	2	°F	0-5 v
Photoimaging Temperature	2	°F	0-5 v
Infrared Scanner Temperature	2	°F	0-5 v
Broad Band IR Spectrometer Temperature	2	°F	0-5 v
Data Automation Equipment	6	status	1 bit
Data Automation Equipment Temperature	1	°F	0-5 v
Power Switching Electronics Temperature	1	°F	0-5 v
Power Switching Electronics	1	status	1 bit
Scan Platform No. 1 Temperature	1	°F	0-5 v
Scan Platform No. 1	1	status	1 bit
Scan Platform No. 2	1	status	1 bit
Scan Platform No. 2, Temperature	1	°F	0-5 v

Table 2-1: SPACECRAFT ENGINEERING MEASUREMENT LIST
(Sheet 3 of 9)

MEASUREMENT			SIGNAL RANGE
SOURCE	QTY.	UNITS	
RADIO SUBSYSTEM			
S-Band Receiver AGC (Coarse)	2	volts	0-5 v
S-Band Receiver AGC (Fine)	2	volts	0-5 v
S-Band Receiver Static Phase Error	2	volts	0-5 v
S-Band Receiver VCO Output Level	2	milliwatts	0-5 v
S-Band Receiver LO Drive Level	2	milliwatts	0-5 v
S-Band Receiver Regulated Voltage Level	2	volts	0-5 v
S-Band Receiver Temperature	2	°F	0-5 v
Exciter RF Power Output	2	milliwatts	0-5 v
Exciter Regulated Voltage Level	2	volts	0-5 v
Exciter Temperature	2	°F	0-5 v
Power Amplifier RF Power Output	2	watts	0-5 v
Power Amplifier Anode Voltage	2	volts	0-5 v
Power Amplifier Helix Current	2	milliamps	0-5 v
Power Amplifier Collector Current	2	milliamps	0-5 v
Power Amplifier Collector Temperature	2	°F	0-5 v
Power Amplifier Converter Temperature	2	°F	0-5 v
Launch Transmitter Temperature	1	°F	0-5 v

Table 2-1: SPACECRAFT ENGINEERING MEASUREMENT LIST
(Sheet 4 of 9)

MEASUREMENT			SIGNAL RANGE
SOURCE	QTY.	UNITS	
RADIO SUBSYSTEM (CONT)			
Relay Receiver AGC	2	volts	0-5v
Relay Receiver Performance	4	volts	0-5v
Relay Receiver Temperature	2	°F	0-5v
Relay Receiver Detector Lock	2	cond	1 bit
VCO Temperature	2	°F	0-5v
Power Sources A or B	1	cond	1 bit
VCO Counter	2	count	10 bits
TELEMETRY SUBSYSTEM			
Vehicle Identification	1	vehicle	2 bits
Temperature	2	°F	0-5v
Reference Voltages	4	volts	0-5v
Power Supply Voltages	4	volts	0-5v
Power Supply Status	8	cond	1 bit
Oscillator Status	1	cond	1 bit
T/M Mode	6	cond	1 bit
Frame Sync	1	sync.	15 bit
Frame Time	1	time	12 bit
GUIDANCE AND CONTROL SUBSYSTEM			
Star Mapping Signal No. 1	1	stellar magnitude	0-5v
Star Mapping Signal No. 2	1	stellar magnitude	0-5v

TABLE 2-1: SPACECRAFT ENGINEERING MEASUREMENT LIST
(Sheet 5 of 9)

MEASUREMENT			SIGNAL RANGE
SOURCE	QTY.	UNITS	
GUIDANCE AND CONTROL SUBSYSTEM (CONT)			
Canopus Roll Error No. 1	1	angle	0-5v
Canopus Roll Error No. 2	1	angle	0-5v
Canopus Recognition No. 1	1	cond	1 bit
Canopus Recognition No. 2	1	cond	1 bit
Limb and Terminator Sensor	2	cond	1 bit
IRU Roll Position	2	angle	0-5v
IRU Pitch Position	2	angle	0-5v
IRU Yaw Position	2	angle	0-5v
IRU Roll Rate	2	angular rate	0-5v
IRU Pitch Rate	2	angular rate	0-5v
IRU Yaw Rate	2	angular rate	0-5v
Roll Spin Motor No. 1	1	sync cond	1 bit
Pitch Spin Motor No. 1	1	sync cond	1 bit
Yaw Spin Motor No. 1	1	sync cond	1 bit
Roll Spin Motor No. 2	1	sync cond	1 bit
Pitch Spin Motor No. 2	1	sync cond	1 bit
Yaw spin Motor No. 2	1	sync cond	1 bit

Table 2-1: SPACECRAFT ENGINEERING MEASUREMENT LIST
(Sheet 6 of 9)

MEASUREMENT			SIGNAL RANGE
SOURCE	QTY.	UNITS	
GUIDANCE AND CONTROL SUBSYSTEM (CONT)			
Roll Failure Detect	2	cond	1 bit
Pitch Failure Detect	2	cond	1 bit
Yaw Failure Detect	2	cond	1 bit
Roll Gyro No. 1, Temperature	1	°F	0 - 5v
Pitch Gyro No. 1, Temperature	1	°F	0 - 5v
Yaw Gyro No. 1, Temperature	1	°F	0 - 5v
Roll Gyro No. 2, Temperature	1	°F	0 - 5v
Pitch Gyro No. 2, Temperature	1	°F	0 - 5v
Yaw Gyro No. 2, Temperature	1	°F	0 - 5v
Sun Acquisition Signal No. 1	1	cond	1 bit
Sun Acquisition Signal No. 2	1	cond	1 bit
Sun Sensor A Pitch Error	1	angle	0 - 5v
Sun Sensor A Yaw Error	1	angle	0 - 5v
Sun Sensor B Pitch Error	1	angle	0 - 5v
Sun Sensor B Yaw Error	1	angle	0 - 5v
Nitrogen Pressure, Manifold	2	psia	0 - 5v
Nitrogen Temperature	4	°F	0 - 5v
TVC Pitch Position	1	angle	0 - 5v
TVC Yaw Position	1	angle	0 - 5v
TVC Actuator Temperature	2	°F	0 - 5v
Thruster Voltage On/Off	16	cond	1 bit

Table 2-1: SPACECRAFT ENGINEERING MEASUREMENT LIST
(Sheet 7 of 9)

MEASUREMENT			SIGNAL RANGE
SOURCE	QTY.	UNITS	
DATA STORAGE SUBSYSTEM			
Recorder Case Temperature	8	°F	0-5v
Recorder Pressure	8	psia	0-5v
Malfunction	8	cond	1 bit
Tape Content	8	inches	7 bits
Record On/Off	8	cond	1 bit
Tape Speed Mode	8	cond	1 bit
End of Tape	8	cond	1 bit
Start of Tape	8	cond	1 bit
COMPUTING & SEQUENCING SUBSYSTEM			
Programmer Data	10	words	27 bits
C&S Ready	1	cond	1 bit
Parity Errors	6	cond	1 bit
Propulsion Rectifier Voltage	1	volts	0-5v
Accelerometer Null Detector	2	cond	1 bit
S/C Time	1	sec	27 bits
Status Signals	31	cond	1 bit
Command Word A	1	cond	1 bit
Command Word B	1	cond	1 bit
Command Status	1	word	27 bits
POWER SUBSYSTEM			
Battery Voltage	3	volts	0-5v
Maneuver Bus Voltage	1	volts	0-5v

Table 2-1: SPACECRAFT ENGINEERING MEASUREMENT LIST
(Sheet 8 of 9)

MEASUREMENT			SIGNAL RANGE
SOURCE	QTY.	UNITS	
POWER SUBSYSTEM (CONT)			
Solar Array Current	1	amps	0-5v
S/C Current	1	amps	0-5v
Unregulated DC Bus Voltage	1	volts	0-5v
Maneuver Bus Voltage	1	volts	0-5v
2400 Hz Inverter Output Voltage	3	volts	0-5v
2400 Hz Inverter Temperature	3	°F	0-5v
2400 Hz Inverter Output Current	3	amps	0-5v
2400 Hz Inverter Output Frequency	3	Hz	0-5v
Battery Charger Output Current	3	amps	0-5v
Battery Charger Temperature	3	°F	0-5v
Solar Panel Temperature	4	°F	0-5v
2400 Hz Inverter Temperature	3	°F	0-5v
Batteries A, B and C Temperature	3	°F	0-5v
Capsule DC Current	2	amps	0-5v
Capsule Voltage	1	volts	0-5v
Solar Gate	1	cond	1 bit
PROPULSION SUBSYSTEM			
Solenoid Valves (engine low thrust)	8	position	1 bit
Solenoid Valves (engine high thrust)	4	position	1 bit
Helium Pressure Transducer	1	psia	0-5v
Helium Temperature Transducer	2	°F	0-5v

Table 2-1: SPACECRAFT ENGINEERING MEASUREMENT LIST
(Sheet 9 of 9)

MEASUREMENT			SIGNAL RANGE
SOURCE	QTY.	UNITS	
PROPULSION TELEMETRY SUBSYSTEM (CONT)			
Fuel Pressure Transducer	1	psia	0-5v
Fuel Temperature Transducer	2	°F	0-5v
Engine Chamber Pressure Transducer	1	psia	0-5v
Oxidizer Pressure Transducer	1	psia	0-5v
Oxidizer Temperature Transducer	2	°F	0-5v
Oxidizer Start Tank Flow Meter	1	lb/sec	0-5v
Fuel Start Tank Flow Meter	1	lb/sec	0-5v
Oxidizer Main Tank Flow Meter	1	lb/sec	0-5v
Fuel Main Tank Flow Meter	1	lb/sec	0-5v
Throttle Actuator Position	1	position	1 bit

The upper subcarrier data consist of orbital science and relayed capsule science. Orbital science data are derived primarily from instruments mounted on two science scan platforms. Orbital science instruments for the 1973 baseline mission include photoimaging vidicons, an infrared radiometer, a high resolution infrared spectrometer, a broadband infrared spectrometer, and an ultraviolet spectrometer.

Lower and upper subcarrier downlink data transmitted by the spacecraft are received by the DSIF. These data are demodulated, decommutated, and decoded at the DSIF, and then transmitted to the SFOF. Data decoding is necessary because of the bi-orthogonal coding technique selected for the upper subcarrier. The types and rates of data quantities received by the DSIF and SFOF are summarized in Figure 2-2.

2.4.2 Types and Quantities of Uplink Data/Commands

Uplink data/commands required by the spacecraft under both routine and contingency conditions are summarized by subsystem in Table 2-2. These commands have the same word format and are transmitted to the spacecraft at the same rate. Each command can require up to 70 bits of information. A command rate of 1 bps was selected as it can be received by the spacecraft's low gain antenna. The resulting command time of 70 seconds is acceptable as it is significantly smaller than the uplink transmission time of more than 10 minutes at nominal encounter distances. This low 1 bps command transmission rate does not impose stringent requirements on the MDE. The total quantity of commands is determined by the mission command profile. A preliminary sequence of commands for a nominal mission is given in Figure 2-3.

2.4.3 Anticipated Real-Time Contingency Actions

Real-time contingencies result from malfunctions or out-of-tolerance conditions in either the spacecraft or the MOS.

Real-time contingency actions due to spacecraft malfunctions are affected strongly by transmission times. At conjunction distances (worst case), round trip transmission times are in excess of 40 minutes. The effect of these long transmission times on the contingency telemetry/command sequence is illustrated in Figure 2-4. A one-way transmission time of 12 minutes was assumed for this example. If a malfunction occurs, 12 minutes will elapse before its occurrence is known on Earth. If the spacecraft command receiver is not locked up, the lockup process will take 34 minutes. This assumes that lockup itself is achieved in 10 minutes and that one-way transmission time is 12 minutes. Once lockup is achieved, another 12 minutes are required for contingency commands to reach the spacecraft. Twelve more minutes are required to transmit the effects of the contingency commands back to Earth. It was assumed here that the 34 minutes required for lockup process were sufficient to prepare contingency commands. Therefore, the time from malfunction to verification of corrective action is 70 minutes.

There are several types of spacecraft contingencies. One type occurs when a scheduled command is not executed. For critical scheduled events, backup commands can be prepared and the command receiver locked up in advance. For the example shown in Figure 2-4, a saving of 34 minutes would result.

SFOF	DSIF
DATA MODE:	
<div> <div>S/C 1 - 75</div> <div>S/C 2 - 75</div> <div>Total 150 bps</div> </div>	1. Cruise
	<div> <div>S/C 1 Lower Subcarrier 75</div> <div>S/C 2 Lower Subcarrier 75</div> <div>Total 150 bps</div> </div>
	2. Orbital Operations (Early Encounter)
	<div> <div>S/C 1 - 300</div> <div>S/C 1 - 48,000</div> <div>S/C 2 - 300</div> <div>S/C 2 - 48,000</div> <div>Total 96,600 bps</div> </div>
	<div> <div>S/C 1 Lower Subcarrier 300</div> <div>S/C 1 Upper Subcarrier 153,600</div> <div>S/C 2 Lower Subcarrier 300</div> <div>S/C 2 Upper Subcarrier 153,600</div> <div>Total 307,800 bps</div> </div>
	3. Orbital Operations (Nominal Encounter)
<div> <div>S/C 1 - 300</div> <div>S/C 1 - 24,000</div> <div>S/C 2 - 300</div> <div>S/C 2 - 24,000</div> <div>Total 48,600 bps</div> </div>	<div> <div>S/C 1 Lower Subcarrier 300</div> <div>S/C 1 Upper Subcarrier 76,800</div> <div>S/C 2 Lower Subcarrier 300</div> <div>S/C 2 Upper Subcarrier 76,800</div> <div>Total 154,200 bps</div> </div>
	4. Orbital Operations (Late Encounter)
	<div> <div>S/C 1 - 300</div> <div>S/C 1 - 12,000</div> <div>S/C 2 - 300</div> <div>S/C 2 - 12,000</div> <div>Total 24,600 bps</div> </div>
	<div> <div>S/C 1 Lower Subcarrier 300</div> <div>S/C 1 Upper Subcarrier 38,400</div> <div>S/C 2 Lower Subcarrier 300</div> <div>S/C 2 Upper Subcarrier 38,400</div> <div>Total 77,400 bps</div> </div>
	5. Orbital Operations (Medium Gain Antenna Backup)
	<div> <div>S/C 1 - 120</div> <div>S/C 1 - 1200</div> <div>S/C 2 - 120</div> <div>S/C 2 - 1200</div> <div>Total 2,640 bps</div> </div>
<div> <div>S/C 1 - 8-1/3</div> <div>S/C 2 - 8-1/3</div> <div>Total 16-2/3 bps</div> </div>	6. Emergency
	<div> <div>S/C 1 Lower Subcarrier 8-1/3</div> <div>S/C 2 Lower Subcarrier 8-1/3</div> <div>Total 16-2/3 bps</div> </div>

Figure 2-2: TYPES AND RATES OF GROUND-RECEIVED DOWN-LINK DATA

Table 2-2: PRELIMINARY UPLINK COMMAND LIST (Sheet 1 of 2)

RADIO SUBSYSTEM

Switch Receiver
 Switch to Medium-Gain Antenna
 Switch to Low-Gain Antenna B
 Switch to Power Amp B
 Switch to Power Amp A
 Switch to Exciter B
 Switch to Exciter A
 Switch to Low-Gain Antenna (A)
 Switch to High-Gain Antenna
 Ranging On/Off
 Launch Transmitter On
 Relay Receivers On
 Relay Receivers Off
 High-Gain Antenna Pointing (2)
 Medium-Gain Antenna Pointing

TELEMETRY (T/M) SUBSYSTEM

T/M Mode 1
 T/M Mode 2
 T/M Mode 3
 T/M Mode 4
 T/M Mode 5
 T/M Mode 6

COMPUTING AND SEQUENCING
SUBSYSTEM

Compare
 Add
 Subtract
 Load Command Register
 Branch
 Execute Discrete
 Module Two Sum
 Telemetry Word Readout
 Stored Program Address
 Terminate
 Transfer
 Load Memory
 Memory Dump
 Velocity - Low Thrust
 Pitch Plus
 Pitch Minus
 Roll Plus
 Roll Minus
 Yaw Plus
 Yaw Minus
 Wait Time

FLIGHT CAPSULE

Capsule Power A On/Off
 Capsule Power B On/Off
 Switch Closure (10)

DATA STORAGE SUBSYSTEM

Mode Control (1 - 18)
 Recorder On/Off (1 - 8)
 Recorder Playback On/Off (1 - 8)

PYROTECHNIC SUBSYSTEM

Pyrotechnic Subsystem On/Off
 Low-Gain Antenna Deployment (2 booms)
 Medium-Gain Antenna Support Release
 High-Gain Antenna Support & Deployment
 VHF Relay Antenna Support
 Science Scan Platform Support & Deployment
 Science Instrument Covers Removal
 Sterilization Canister Release
 Flight Capsule Umbilical Release
 Flight Capsule Separation
 Midcourse Correction 1 Squib Valve
 Midcourse Correction 2 Squib Valves
 Midcourse Correction 3 Squib Valves
 Orbit Insertion Squib Valves
 Orbit Trim Squib Valves
 Planetary Vehicle Separation
 Reaction Control N₂ A and B Squib Valves

THERMAL CONTROL SUBSYSTEM

Heater On/Off Control (16)

PROPULSION SUBSYSTEM

Pressurant Line Squib Valves Open/Close
 Start Tank Feedline Squib Valves Open/Close
 Start Tank Feedline Solenoid Valves
 Open/Close
 Main Tank Feedline Squib Valves Open/Close
 Engine Flow Control Valves Open/Close
 Engine Throttle Position High/Low
 Main Tank Feedline Solenoid Valves
 Open/Close
 Propulsion Subsystem Power On/Off

GUIDANCE AND CONTROL SUBSYSTEM

Attitude Control Electronics A/B Select
 N₂ Isolation A Open/Close
 N₂ Isolation B Open/Close
 N₂ Crossover Open/Close
 Accelerometer On/Off
 Attitude Stabilize
 Celestial Reference
 Roll Inertial Hold
 Pitch Inertial Hold
 Yaw Inertial Hold
 Maneuver Plus/Minus
 Roll Slew On/Off
 Pitch Slew On/Off
 Yaw Slew On/Off
 TVC DC On/Off
 TVC AC On/Off
 TVC Gain High/Low
 Canopus Tracker A On/Off
 Canopus Tracker B On/Off
 Select Canopus Tracker A/B
 Canopus Cone Angle Update
 Canopus Upper Gate Disable (Units 1 & 2)
 Limb/Terminator Sensor On/Off
 Pitch Coarse Sun Sensor Disable
 Yaw Coarse Sun Sensor Disable

Table 2-2: PRELIMINARY UPLINK COMMAND LIST (Sheet 2 of 2)

IRU 1 Heater On/Off
 IRU 2 Heater On/Off
 IRU 1 On/Off (3 Commands: Pitch, Roll, Yaw)
 IRU 2 On/Off (3 Commands: Pitch, Roll, Yaw)
 IRU Roll Mode Select
 IRU Pitch Mode Select
 IRU Yaw Mode Select
 Select Roll 1 — 2 (COM)
 Select Pitch 1 — 2 (COM)
 Select Yaw 1 — 2 (COM)

SCIENCE SUBSYSTEM

T.V. Select (3 cameras)

Per Camera:

On
 Off
 Focus Step
 Shutter Setting
 Shutter Setting
 Filter Selection
 Filter Selection
 Mode Selection
 Mode Selection
 Mode Selection
 Gain Selection
 Readout (R/O) On
 Readout (R/O) Off

UV OR IR SPECTROMETERS

On
 Off
 Slit Step
 Scan Rate
 Scan Rate
 Readout On
 Readout Off
 Cover Open
 Cover Closed
 Mode Select
 Mode Select
 Mode Select
 Calibrate

IR BROAD BAND SPECTROMETER

Science Readout
 Engineering Readout
 Channel Select
 Channel Select
 Science Multiplex
 Engineering Multiplex
 On
 Off
 Calibrate

IR RADIOMETER

On
 Off
 Readout On
 Readout Off
 Calibrate
 Gain Step

SCAN PLATFORMS (TWO)

Pitch +
 Pitch -
 Yaw +
 Yaw -
 Roll +
 Roll -
 Platform Select

ATMOSPHERIC POLARIMETER

Reduce Scan Limits
 Power Reduce
 Read Out
 Power On/Off

PLASMA PROBE

Change Program Cycle
 Power On/Off

COSMIC RAY TELESCOPE

Calibrate
 Power On/Off

TRAPPED RADIATION DETECTOR ON/OFF

ION CHAMBER ON/OFF

ATMOSPHERE MASS SPECTROMETER ON/OFF

COSMIC DUST DETECTOR

Calibrate
 Power On/Off

POWER SUBSYSTEM

Battery Charger A Inhibit
 Normal Charge Current A
 Trickle Charge Current A
 Increase Charge Current A
 Battery Charger B Inhibit
 Normal Charge Current B
 Trickle Charge Current B
 Increase Charge Current B
 Battery Charger C Inhibit
 Normal Charge Current C
 Trickle Charge Current C
 Increase Charge Current C
 Fail Sense Bus A Reset
 Fail Sense Bus A Trip
 Fail Sense Bus B Reset
 Fail Sense Bus B Trip
 Fail Sense Bus C Reset
 Fail Sense Bus C Trip
 Share Sensor & Boost Inhibit
 Solar Gate Override
 Share Sensor Reset

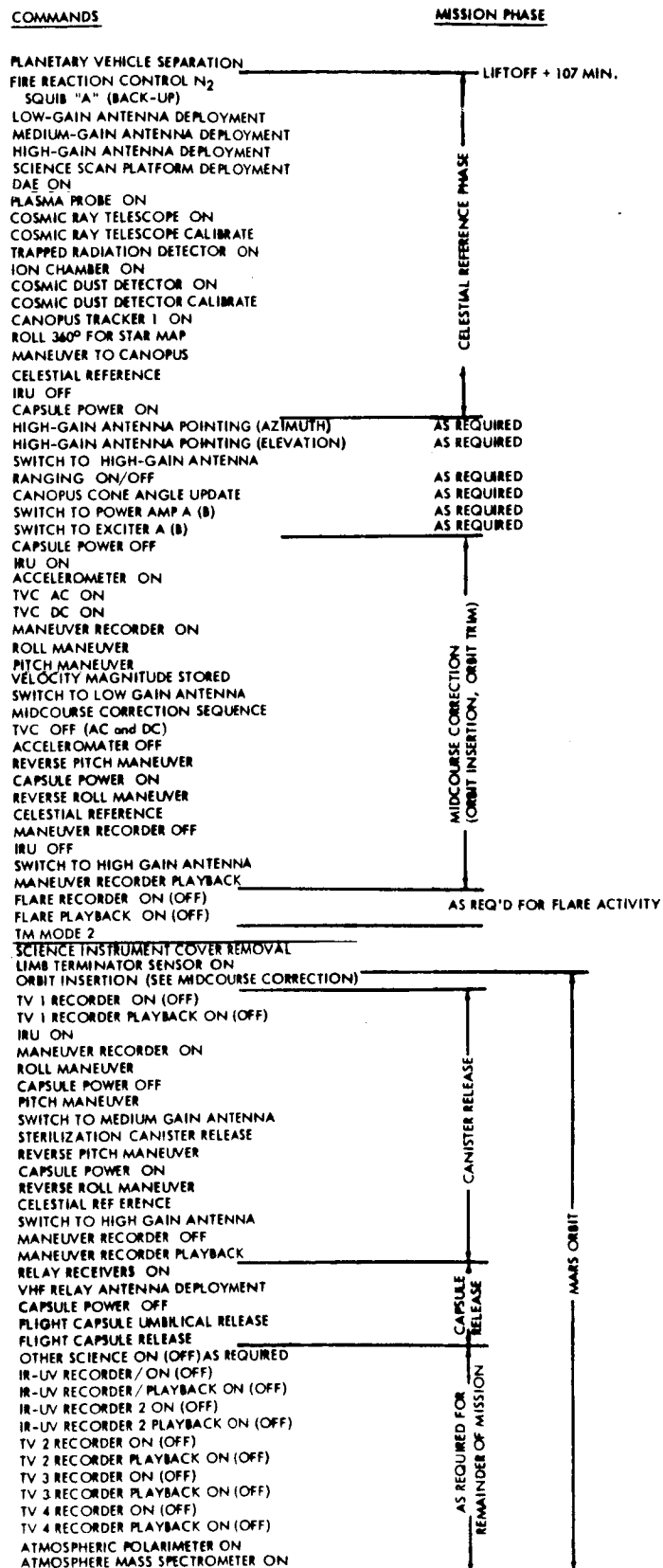


Figure 2-3: TYPICAL MISSION COMMAND PROFILE

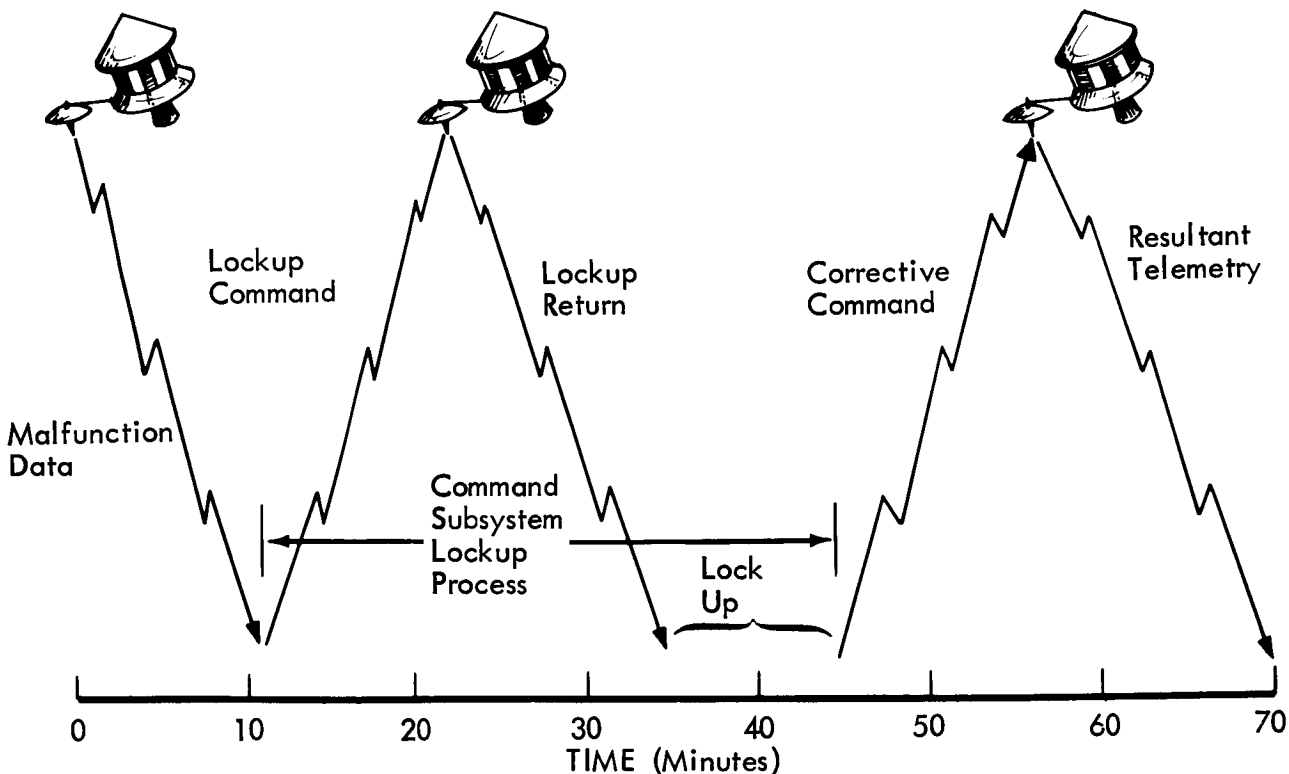


Figure 2-4: CONTINGENCY COMMAND TIME SEQUENCE

An example of such a precomputed command is the use of a backup velocity command for orbit insertion. Such a command would be precomputed, but transmitted only after examination of spacecraft telemetry data establishes that (1) the correct insertion attitude was attained, and (2) the LMD engine did not ignite. In the event this backup command succeeded in igniting the engine, orbit insertion would commence less than $\frac{1}{2}$ hour later than originally scheduled (24 minutes of transmission time and 6 minutes to accumulate several telemetry frames and make a decision). This is a marginal case. A delay of more than $\frac{1}{2}$ hour in engine ignition will inhibit orbit attainment.

A second type of spacecraft contingency to be considered is one that can be predicted by extrapolating telemetry data. For example, a spacecraft attitude maneuver may be commanded to correct for an incipient overhear condition in a critical subsystem. Such incipient conditions can be anticipated with the spacecraft simulation program. (See Section 2.4.5 below.) Random malfunction is a third category of spacecraft contingencies. Here, diagnostic MDE software is used.

As an aid in determining the course of action to be taken in any spacecraft contingency, one ground test spacecraft should be made available during the mission. This spacecraft will be used to simulate the mission spacecraft conditions. It also will serve as a "test bed" upon which to check out contingency solutions. This test spacecraft is not considered an MDE item.

Ground contingencies result from failures within the MDE or the DSN. MDE failures can be minimized through redundancy in critical components.

Mission-dependent software can be used in the event of DSN contingencies such as a partial failure in the data transmission link between a DSIF site and the SFOF. Edit modes in the DSIF computer are used for selecting key telemetry data to be sent to the SFOF in real time.

2.4.4 Mission-Dependent Hardware

The data presented in the foregoing sections were used to develop MDE hardware requirements. Table 2-3 tabulates these requirements for the 1973 Voyager mission, assuming current MDE functional requirements. Redundant equipment has been indicated where loss of the item would mean loss of real-time spacecraft data.

Table 2-3: MISSION-DEPENDENT HARDWARE

Item	Number Required	Number Redundant	Number Spares	TOTALS		
				Number Required per DSIF	Number Required per SFOF	Total Number Required
*Upper Subcarrier Demod. (and filter)	2	2	1	5	0	20
*Upper Subcarrier Synchronizer	2	2	1	5	0	20
A/D Converter	2	2	1	5	0	20
Block Decoder	2	2	1	5	0	20
*Recorder	2	2	1	5	0	20
Filter (2-Channel Demod)	2	2	1	5	0	20
*Lower Subcarrier Demod (and Filter)	2	2	1	5	0	20
*Lower Subcarrier Synchronizer	2	2	1	5	0	20
*Buffer & Formatter	2	2	1	5	0	20
Tape Assembly	1	0	0	1	0	4
Test Selector	1	0	0	1	0	4
Block Comparator	1	0	0	1	0	4
Test Patch Panel	1	0	0	1	0	4
*Data Printer	2	0	1	3	0	12
Control Panel	2	0	1	3	0	12
Real Time Alarm	2	0	1	0	3	3
Current Spacecraft & Telemetry Status Display	2	0	1	0	3	3

*Jet Propulsion Laboratory's Engineering Planning Document No. 283, Revision 2 implies that these items will be replaced by mission-independent equipment for the Voyager mission.

The block diagram in Figure 2-5 shows the major MDE hardware elements required for one spacecraft at the DSIF. MDE/DSIF interfaces also are indicated.

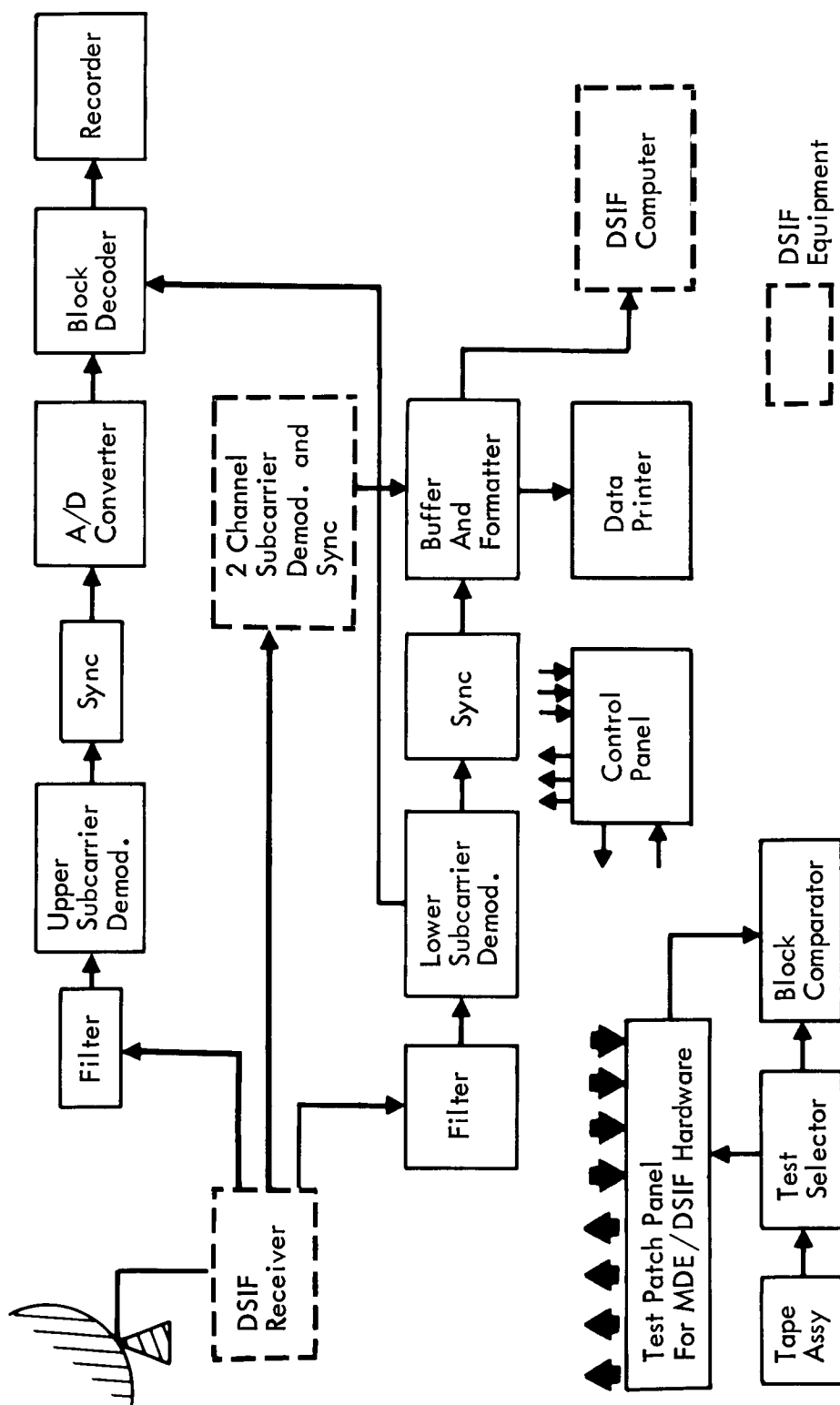


Figure 2-5: BLOCK DIAGRAM FOR KEY MDE HARDWARE AT THE DSIF

The functional requirements of the principal units shown in Figure 2-5 are described in the following paragraphs.

The principal MDE hardware is required at the Deep Space Instrumentation Facility for demodulating and decommutating the telemetry stream. A filter is used for upper subcarrier demodulation and decommutation (upper row of blocks). It separates the two upper channel frequencies present in the detected output of the DSIF receiver. The upper subcarrier demodulator furnishes a data stream to the synchronizer. The synchronizer develops and provides bit sync and data to the analog-to-digital (A/D) converter. The A/D converter tags each bit with a weighting number. This weighting number, which ranges from 0 to 64, is proportional to the output voltage of the demodulator's bit integrator. It indicates the equipment's "confidence" in the correctness of the bit decision ("1" or "0"). The block decoder accepts (1) input data stream and weighting data from the A/D converter, and (2) block sync data from the lower subcarrier demodulator. It decodes the biorthogonally encoded data. These decoded data are then recorded, or, if the proposed DSN 4.5 MHz video is available for Voyager, relayed in real time to SFOF.

The two-channel subcarrier demodulator and synchronizer is mission-independent equipment used to develop data, word sync, and bit sync for the lower subcarrier data in telemetry mode 6 only.

The filter associated with the lower subcarrier demodulation and decommutation equipment separates the three lower channel frequencies. The lower subcarrier demodulator regenerates the subcarrier frequency. The subcarrier frequency is then block-synchronized by the block decoder. The lower subcarrier demodulator also furnishes a data train to the synchronizer which develops bit sync and conditions the data train. The buffer-and-formatter develops and supplies frame sync and data to the DSIF computer and displays select real-time spacecraft data.

The test patch panel allows critical points in the above equipment to be monitored. It also routes taped test signals to check the DSIF telemetry equipment (e.g., simulated downlink signal). The block comparator checks the output of the block/decoder and the buffer-and-formatter against the test data.

A data printer is available for each spacecraft at each DSIF. This printer will print out in real time the lower subcarrier channel numbers and associated values.

A control panel is required for switching in redundant units and selecting inputs from the DSIF receiver, playback from the recorders, or test input signals.

One real-time alarm per spacecraft will exist at the SFOF. This equipment will permit display of any out-of-tolerance condition, as determined by software examination of spacecraft engineering data.

A current spacecraft status and current telemetry status display will be required for each spacecraft at the SFOF. These two displays would be identical were it not for the telemetry data transmission delay. The telemetry status display is driven by incoming telemetry data. The spacecraft status display is driven by the spacecraft simulation software. The two displays will be arranged to facilitate comparison.

2.4.5 Mission-Dependent Software

The following major areas require software development:

- 1) Telemetry and command data handling (TCD)
- 2) Flight path analysis and command (FPAC)
- 3) Spacecraft performance analysis and command (SPAC)
- 4) Mission integration and control (MIC)
- 5) Simulation for software and system checkout and operational training.

Table 2-4 is a summary of the Voyager mission software requirements. The table indicates which programs are expected to be mission-independent for the 1973 Voyager mission, which are mission-dependent, and which are obtained by making mission-peculiar modifications to mission-independent programs. The projected status of these programs is based on current DSN capabilities and the planned software status as defined in Jet Propulsion Laboratory's Engineer Planning Document No. 283, Revision 2. A detailed discussion of these major software areas follows.

Telemetry and Command Data Handling (TCD) -- The telemetry and command data handling software will consist of an integrated set of computer programs for both DSIF and the SFOF. The primary functions of these programs are to process the incoming telemetry originating at the spacecraft, and process the command signals for transmission to the spacecraft. In addition, provisions must be made for an emergency backup processing capability at the DSIF in the event of data link failure or SFOF equipment failure.

The TCD computer software requirements are based on past spacecraft experience and Voyager functions.

The TCD includes four major software subsystems as follows:

- 1) DSIF Computer Command Processing Subsystem -- This subsystem provides the capability for displaying, storing, and controlling transmission of commands from the SFOF to the spacecraft via the DSIF. (It is assumed that the command verification process will be accomplished by mission-independent software.)

The command transmission processor feeds the commands to the DSIF transmitting equipment for transmission to the Voyager spacecraft. After the command has been transmitted, a request is sent to the computer for the next command. This process is repeated until the complete command string has been transferred to the spacecraft computer and sequencer.

- 2) DSIF Computer Telemetry Data Handling Subsystem -- This subsystem provides for (1) buffering, formatting, and editing of the telemetry data stream as it is received from the on-site ground equipment, and (2) controlling the transmission of the telemetry data to the SFOF.

Table 2-4: MISSION SOFTWARE REQUIREMENTS

Software Nomenclature	Function Performed	Where Used		Software Development		
		SFOF	DSIF	Mission Independent (80 to 100% Existing)	Modified Mission Independent	Mission Dependent (80 to 100% New)
Telemetry and Command Data Handling (TCD)	DSIF Command Processing		X			X
	DSIF Command Verification		X	X		
	DSIF Telemetry Data Handling		X			
	Input Processor					X
	Format Edit					X
	Output Processor			X		
SFOF Real-Time Telemetry Processing	Input Processor	X				
	Telemetry Processor			X		
	Output Processor				X	
					X	
	SFOF Telemetry and Tracking Data Processor	X				
	Input Processor			X		
Flight Path Analysis and Command (FPAC)	Orbit Determination	X				X
	Guidance and Maneuver Analysis	X			X	
	Mission Information	X				X
Spacecraft Performance Analysis and Command (SPAC)	System Monitor and Control	X				X
	Contingency Aids	X				X
Mission Integration and Control (MIC)	Master Timing	X			X	
	Command Generation	X			X	
	Mission Events	X			X	
	Data Bank	X			X	
Simulation for Training and Software Checkout	Simulated Real-Time Telemetry Generation	X				X
	Simulated Real-Time Tracking Generation	X				X

Three major software programs are needed to accomplish the above requirements. The first program is the "Input Processor." This program accepts the data words from the telemetry decommutator and transfers them into a memory buffer.

The second program, "Telemetry Processor Format and Edit" formats and edits raw telemetry data from the input buffer. Prior to transmission, edit modes are required in the event the data transmission system becomes partially disabled. When that happens, only a selected portion of the telemetry measurements are sent because of the lower transmission rate available. Several edit modes will be required to cover the various DSN communication failure modes for the various mission phases.

The third program, "Output Processor," prepares the formatted and raw telemetry data from the output buffer for transmission to the SFOF. This program will tag each telemetry frame with spacecraft identification.

- 3) SFOF Real-Time Telemetry Processing Subsystem -- This subsystem processes the telemetry data as first received by the SFOF. The received telemetry measurements are displayed by teleprinter, high speed printer, or plotter. Consequently, the received telemetry data must be edited, formed into telemetry frames for decommutation, and routed to the display devices. In addition, alarm monitoring is performed on selected telemetry measurements.

Three major computer programs are required to perform these functions. The first program "Input Processor," consists of existing mission-independent routines which route the data for further processing. The second program, "Telemetry Processor," forms complete telemetry frames and stores the decommutated data into the telemetry input buffer. In addition, the telemetry processor performs the alarm monitoring function by flagging out-of-tolerance data. The third program, "Output Processor," also consists of existing mission-independent routines. These routines require modification for compatibility with the Voyager mission. These routines provide for displaying the raw telemetry for selected telemetry measurements in real time.

- 4) SFOF Telemetry and Tracking Data Processor Subsystem -- This subsystem processes telemetry data for subsequent use by analysis programs and formats stored telemetry data for display similar to the real-time displays.

The data processor subsystem consists of three main computer programs. The first program, "Input Processor," examines the raw incoming telemetry data and routes the information for further processing (e.g., routing of tracking data to FPAC). The second program, "Telemetry Processor," is a mission-dependent program. This program examines raw telemetry data that contains identification words and formats the raw data into telemetry frames. It then time-tags the frames and decommutates them. It also flags measurements with data parity error. The third program prepares requested past-time data displays from data stored on the master data tables. This third program is an existing mission-independent program that will require modification.

Flight Path Analysis and Command (FPAC) -- The FPAC system software provides programs for orbit determination, guidance and maneuver analysis, and mission information. The software will be programmed for the SFOF computer and associated peripheral equipment. The following is a list of the major functions of the FPAC programs:

1) Orbit Determination

- a) Compute the spacecraft trans-Mars trajectory orbital elements, and associated errors, based upon nominal trajectory information and DSIF tracking data.
- b) Reestablish the vehicle's trajectory after orbit insertion and trim, and verify compliance with planetary quarantine requirements.
- c) Determine the significant Mars harmonic coefficients.

2) Guidance and Maneuver Analysis

- a) Compute optimum Mars orbit insertion conditions to minimize insertion velocity requirements for given trajectory conditions.
- b) Perform a rapid preliminary trajectory search using simple conic computation. This is required in the event of large errors.
- c) Perform a precise midcourse trajectory search to find a trajectory that satisfies the optimum Mars orbit insertion parameters. This determines the required midcourse maneuver.
- d) Predict the spacecraft's maneuver errors, combine these errors with orbit prediction errors, and map the total error volume through normal transfer to Mars orbit insertion.
- e) Compute the pitch, yaw, and roll magnitude to effect the required attitude for midcourse correction maneuver.
- f) Compute the Mars orbit insertion maneuver that ensures the minimum insertion velocity maneuver consistent with desired Mars orbit.
- g) Compute the thrust orientation and duration required to achieve the selected insertion maneuver, taking into account the effects of finite thrust time.
- h) Compute the pitch, yaw, and roll maneuver required to effect the computed thrust orientation for orbit insertion, taking into account the spacecraft maneuver constraints.
- i) Generate the error envelope associated with any proposed maneuver.
- j) Perform orbit trim search, using approximate but fast orbit computation procedures, to find a trimmed orbit that satisfies the scientific objectives and planetary quarantine requirements.

3) Mission Information (Trajectory)

- a) Calculate predicted spacecraft position and velocities.
- b) Predict the doppler change associated with a successful midcourse maneuver.
- c) Reestablish the spacecraft's trajectory, after midcourse correction, using the old spacecraft position and new DSIF tracking data.
- d) Verify that antenna null regions are not entered during the maneuver.

Spacecraft Performance Analysis and Command (SPAC) -- The basic requirements for the SPAC software are to (1) provide monitoring and control of spacecraft subsystems and (2) provide computer approaches to the solution of anticipated contingencies.

The following functional requirements are to be included in the SPAC software in order to monitor and control each spacecraft subsystem:

- 1) Perform the analyses required to assess subsystem performance and provide status information.
- 2) Predict future subsystem capabilities and status as a function of mission time and event sequence.
- 3) Format output data for specified display devices to the requested formats.
- 4) Accept, in accordance with preestablished priorities, callup from user area input/output consoles.
- 5) Store on disk, or on magnetic tape, specified program outputs for callup by other programs.
- 6) Accept manual input data from punched cards or message composer in lieu of, or in addition to, data from the master data or user program files.
- 7) Generate commands from a program which (1) converts symbolic computer and sequencer (C&S) commands to binary coded words, (2) sets identification and parity bits, and (3) provides listings of the binary sequence and diagnostic comments. In addition, format binary commands for transmission and print commands for a visual inspection.
- 8) Generate a mission sequence of events, and provide a complete integrated schedule of operational activities and significant spacecraft events.
- 9) Maintain a register which contains the constantly updated SFOF - spacecraft command delay time.

The SPAC software also provides programs to aid in the solution of contingency situations. The following software capabilities are to be included as a part of the SPAC software:

- 1) Trend Analysis Programs -- These programs provide a tool with which the subsystem engineer can monitor and predict the performance of a particular subsystem during the mission. The program performs two functions: monitoring of past performance and prediction of future performance. The status phase utilizes spacecraft telemetry to provide a record of past events. The prediction phase uses the mission profile, stored nominal design data, and inputs from other subsystem trend programs. It predicts the future performance of a particular subsystem for a given sequence of events. Status and prediction phase parameters are both printed and plotted.
- 2) Spacecraft Simulation Program -- This program will be an extension of the real-time telemetry generation program described in the checkout and training simulation portion of Section 2.4.5. The purpose of this program is to predict spacecraft status, provide a vehicle for trying proposed command sequences, and provide a means of carrying the spacecraft status ahead a requested amount of time. This program will simulate the spacecraft to the extent required to generate a telemetry stream consistent with this purpose. This software will estimate present and future spacecraft status based on received telemetry data, commands that have been sent or are intended for the time period of interest, and trend data from the subsystem trend analysis program described above. The principal portion of this program is a computer and sequencer simulator which simulates the operation of the C&S subsystem by maintaining the current C&S core map, executing commands as scheduled, and maintaining the simulated clock in phase with the spacecraft clock.

This simulator supplies the data required by the current spacecraft status display described in Section 2.4.4. When a future spacecraft status prediction is requested, it will be displayed in the place of the current status.

- 3) Diagnostic Program -- Upon receipt of out-of-tolerance condition, this program examines stored telemetry data and prints out probable causes as an aid to analysis.

Mission Integration and Control (MIC) -- The requirements establishing the mission integration and control software are based on the need for common information in several areas and assurance that the requirements and actions of each area are compatible with spacecraft design and the mission objectives. Software control is necessary for monitoring and receiving data simultaneously from two spacecraft vehicles and two capsules. The following are the basic requirements for the MIC.

- 1) Ability to correlate the spacecraft clock time with GMT, and to detect errors in timing that may have occurred in the spacecraft.
- 2) Assurance that commands transmitted to the spacecraft are in the correct format and in the planned sequence. Software is required for converting the inputs received in engineering units from both FPAC and SPAC to command word format and sequencing.
- 3) Provide a means of updating pre-event countdown.

- 4) Provide the user programs with all items of parametric data for their operation. Examples of parametric data are the geophysical statistics of the DSIF stations and the gravitational properties of Mars. Three connecting programs are required for providing the above data:
 - a) A utility program to perform initial loading and housekeeping functions independently of other user programs.
 - b) An input program to be employed by a user program in reading data from the file.
 - c) A routine for taking parametric results generated by a user program and storing them for entry in the common environmental data bank.

Simulation for Operational Training, and Software and System Checkout -- A need exists for the real-time generation of simulated spacecraft telemetry and tracking data for the training of operations personnel. The simulated data should be introduced into the system through the DSIF. Two programs are needed to meet the above requirements.

The first is a real-time telemetry generation program. This program must supply realistic telemetry responses to commands, externally supplied environmental data, and preplanned anomalies. The following must be included in the generation of telemetry data:

- 1) Spacecraft computer and sequencer simulation
- 2) Command handling
- 3) Spacecraft subsystem command response generation
- 4) Telemetry data formatting
- 5) Telemetry mode timing
- 6) Option to simulate a spacecraft only or the spacecraft and a DSIF.

The second program performs the real-time tracking data generation function. This program uses modified FPAC software to generate simulated tracking data.

Finally, there is a requirement that the software developed for TCD, FPAC, SPAC, and MIC be checked out both separately and as a total system. The programs developed for simulation training can be utilized to accomplish the major portion of the checkout function.

2.5 CONCLUSION AND RECOMMENDATIONS

The basic mission-dependent equipment requirements for the Voyager mission are similar to those of previous missions using the DSN. Up- and downlink data rates are not expected to create unusual MDE requirements. However, some features of the MDE requirements are unique to Voyager because of first Mars orbiting operations and simultaneous operation of two spacecraft.

Specific MDE recommendations are:

- 1) Spacecraft equipment simulation software should be developed that will permit prediction of spacecraft status and furnish a "test bed" upon which mission operations personnel can evaluate and check out commands.
- 2) There should be separate display and command consoles in the SFOF for each spacecraft.
- 3) There should be a display which facilitates comparison between computed current spacecraft status and current telemetry status.

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3.0 VOYAGER PROGRAM TEST FLIGHT

3.1 OBJECTIVES

The objective of this study was to determine the benefits of a test flight to the Voyager program, using the Saturn IB, in terms of effect on ground testing, level of assurance provided, and effect on cost and facilities.

3.2 SCOPE

The complete Voyager mission environment cannot be fully simulated on the ground. Consequently, a Voyager program flight test appears attractive.

System elements considered for the test flight included both a flight spacecraft and a planetary vehicle. Mission profiles considered for the test flight included an Earth orbit and a deep space flight.

Comparison criteria between a flight test and a ground test included schedule considerations as well as cost and confidence levels. This is because a test flight must support the 1973 mission. Costs associated with mission operations (communications and tracking) were not included in the comparison because of lack of data.

3.3 APPROACH

The approach to this study is shown in Figure 3-1. The study was accomplished in three tasks. A discussion of these tasks is given below.

Task No. 1 -- A set of test requirements was defined. This was accomplished through an analysis of the 1973 mission. The key events in the mission were identified. The key events are those where spacecraft subsystem performance is critical to accomplishing a significant maneuver or function. The subsystems or components associated with key events were identified. These subsystems were evaluated to determine whether a flight test or ground test would result in a higher level of confidence. This evaluation resulted in a list of requirements for flight tests that increased confidence. The impact of such flight tests on the ground test program also was established.

Task No. 2 -- The constraints imposed on the flight test by the Saturn IB capability were considered. For a range of test payloads, the capabilities of the Saturn IB and spacecraft propulsion subsystem for achieving circular, elliptical, and synchronous Earth orbits, as well as a deep space flight, were evaluated. From this evaluation and the previously established flight test requirements, a set of candidate test flights was defined.

Task No. 3 -- The candidate test flights were then evaluated against the following:

- The degree to which all test requirements were satisfied
- The degree to which ground testing was reduced
- The degree to which the use of ground test facilities was reduced

- The level of assurance provided by the addition of a test flight
- The added costs for a flight test
- The impact of a test flight on the 1973 program schedule.

On the basis of the above evaluations, a preferred test flight was selected.

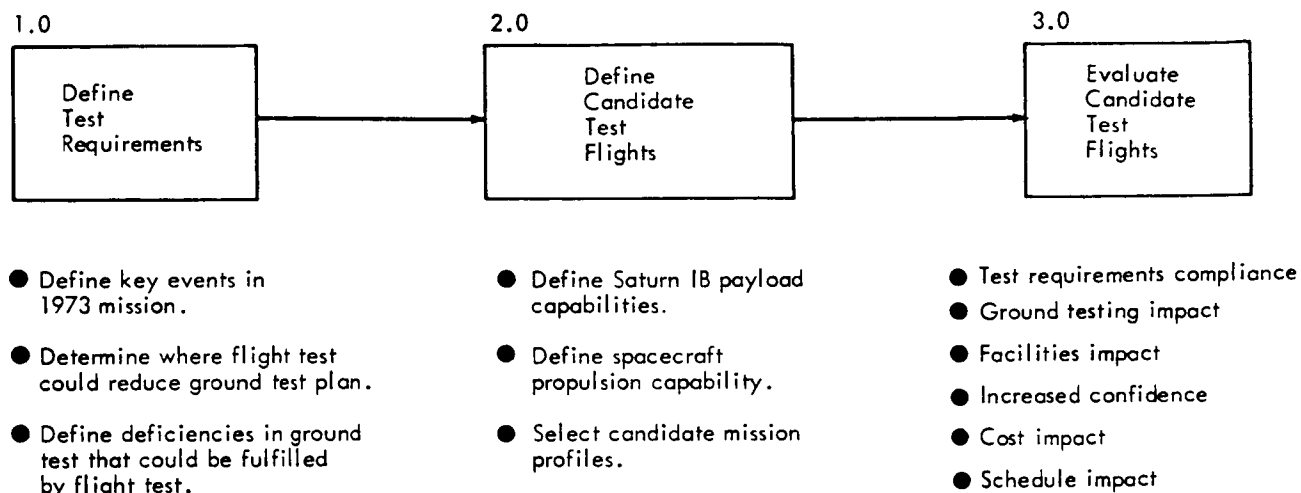


Figure 3-1: FLIGHT TEST STUDY PLAN

3.4 RESULTS

Study results are presented in three sections: (1) flight test requirements, (2) candidate test flight profiles, and (3) evaluation of candidate test flights.

3.4.1 Flight Test Requirements

Flight testing, if performed, should demonstrate the following:

- 1) Payload structural and thermal performance during launch
- 2) Payload separation and sequencing
- 3) Capsule deorbit and entry
- 4) Zero-g deployment
- 5) Propulsion/attitude control/structure interactions

- 6) Adequacy of thermal design in space
- 7) Long life
- 8) Range transponder design adequacy
- 9) Mission operations
- 10) Propulsion subsystem performance
- 11) Canopus star tracker performance
- 12) Science instruments calibration.

The background of each of the above flight test requirements is given below:

- 1) Payload Structural and Thermal Performance During Launch -- Launch environment parameters that influence the structural and thermal performance of the payload are (1) liftoff noise, (2) vibrations, (3) transonic loads, (4) temperature variations, and (5) internal pressure profile.

The launch environment associated with the Saturn V/Voyager space vehicle is influenced by the unique Voyager shroud configuration. A Saturn IB test will use the Voyager shroud and can duplicate closely the Saturn V/Voyager launch environment. This offers the possibility of flight-qualifying the structural and thermal design of the Voyager payload. Such a test flight will substitute for some ground testing.

- 2) Payload Separation and Sequencing -- The requirement for launching two Voyager payloads on one Saturn V launch vehicle results in a complex separation sequence. There is no ground test that can adequately simulate the separation sequence. The dynamics of separation--clearance and trajectory--can only be demonstrated in flight. Such inflight separation offers considerable confidence in a critical phase of the 1973 mission.

- 3) Capsule Deorbit and Entry -- A ground facility for full-scale simulation of Mars entry is not available. A test flight offers the opportunity of demonstrating both full scale capsule entry and deorbit equipment. A deorbit maneuver to place the capsule on an entry trajectory following capsule separation from the spacecraft provides confidence in the programmed sequence, operation of the retropackage, and performance of attitude control functions in a dynamic space environment. The entry maneuver will provide confidence in deployment of entry equipment, performance of the heat shield and aeroshell, and terminal separation. The demonstration of the terminal landing equipment (i.e., parachutes or vernier propulsion) can be performed in the altitude range of 150,000 to 200,000 feet where atmospheric conditions are comparable to those on the Martian surface. Such a test flight could complement or supplement planned capsule development tests.

- 4) Zero-g Deployment -- The spacecraft includes components that must be deployed and operated in a zero-g environment (e.g., antennas). An adequate ground test of zero-g deployment is not feasible.

- 5) Propulsion/Attitude Control/Structural Interaction -- Engine firings for maneuvers result in complex interactions. These interactions involve the structural response and feedback to the gyros, response of the autopilot, and the frequency response and phase lag of the thrust vector control (TVC). The usual procedure by which these interactions are evaluated entails a series of subsystem tests and an analysis integrating the results of such tests, and a captive system level test in a simulated space environment. The subsystem test approach does not provide a high confidence level. A captive system test does not increase the confidence level significantly. Moreover, for a spacecraft the size of Voyager, a captive system test may not be feasible. Such a test requires a reinforced structure to withstand thrust loads in a constrained condition (as opposed to only inertial loads in flight), a degree of restraint to withstand the several seconds burn time to overcome initial transients, and a large vacuum chamber which can maintain a vacuum level during engine firing.

Inflight testing would, on the other hand, provide very high confidence in the stability of the spacecraft control loop. A flight test is more realistic. Further, heating rate data from both engine soak-back and nozzle radiation are also obtained during such a test. Such thermal information would be useful in analyzing thermal design.

- 6) Adequacy of Thermal Design in Space -- Available Mariner space flight data indicate that thermal vacuum tests cannot uncover all thermal design inadequacies. This is due primarily to inadequate space simulation techniques. A test flight would overcome this difficulty.
- 7) Long Life -- The Voyager system must operate in a deep space environment for a prolonged period of time. Full mission simulation of such an environment is costly and impractical. A test flight extending over several months of operation provides a continuous exposure of the spacecraft to a space environment.
- 8) Range Transponder Design Adequacy -- The "regeneration technique" proposed for determining distance and velocity of the Voyager spacecraft is new. In this technique, a signal sent from the DSIF is regenerated by the transponder and returned to the DSIF with a calibrated delay. A transponder ground test is feasible. However, large distance and relative motion simulation is not feasible. A test flight, on the other hand, affords the opportunity to verify transponder design, check the calibration, evaluate Doppler effects, and establish confidence in its reliability.
- 9) Mission Operations -- A test flight utilizing the DSN provides opportunity for exercising the procedures, computer programs, personnel and operational procedures for control of a Voyager mission. In particular, the techniques of mission control and data processing with the communication delay inherent in a deep space mission, can be verified.

For an Earth orbit flight, the degree to which the 1973 mission operations system can be exercised is reduced. The time period during each orbit when the ground stations can all see the spacecraft are limited for low Earth orbit operations.

- 10) Propulsion Subsystem Performance -- Confidence in propulsion subsystem performance will be enhanced by a test flight. Propellant control and sloshing in a

zero-g environment can be overcome by design. A test flight can provide considerable confidence in the effectiveness of that design through demonstration of the subsystem. For example, the rate of settling during the start cycle can be measured, and the stability of the spacecraft with partial tank loading can be determined.

- 11) Canopus Star Tracker Performance -- A key problem of a spacecraft guidance and control subsystem using a star tracker for orientation is that of glint. It is very difficult to simulate true space light conditions on the ground. A test flight provides a realistic test of the star tracker and threshold control for greater confidence in the 1973 Mars mission.
- 12) Science Instruments Calibration -- A test flight provides a possibility of calibrating science instruments against a known environment (Earth's). For example, the UV spectrometer can be calibrated against the known Earth's upper atmosphere.

The above test requirements, with the exception of those for capsule deorbit and entry, do not call for an operational flight capsule.

3.4.2 Candidate Test Flight Profiles

In selecting candidate test flights, four payload configurations were examined: (1) a fully fueled planetary vehicle, (2) a planetary vehicle with a 10% spacecraft fuel load, (3) a fully fueled spacecraft, and (4) a spacecraft with a 10% fuel load. The capability of the Saturn IB for placing these four payloads into circular and elliptic Earth orbits are summarized in Table 3-1. For the circular orbit, the

Table 3-1: VOYAGER/SATURN IB FLIGHT TEST CAPABILITIES

CONFIGURATION AND WEIGHT IN ORBIT	MAXIMUM CIRCULAR ORBIT ALTITUDE (n mi)	MAXIMUM ELLIPTIC ORBIT APOGEE (100 n mi PERIGEE) (n mi)
1 PLANETARY VEHICLE FULL FUEL LOAD	1020	2750
1 PLANETARY VEHICLE 10% FUEL LOAD	2300	7650
SPACECRAFT ONLY (NO CAPSULE) FULL FUEL LOAD	1630	4800
SPACECRAFT ONLY (NO CAPSULE) 10% FUEL LOAD	3490	15120

- Nose Shroud: 4880 lb, 3860 lb Jettisoned at 100 n mi
- Performance based on due east launch, SA-212 vehicle
- S-IVB Assumed Restartable

maximum altitude is given. For the elliptic orbit, the maximum apogee for a 100 nmi perigee is quoted.

The spacecraft propulsion subsystem can be used to augment the payload capability of the Saturn IB. The launch of a fully fueled planetary vehicle can place a capsule in Earth orbit and a spacecraft in either an inclined synchronous Earth orbit, or a deep space trajectory which encounters the orbit of Mars. The above orbits are achieved by first attaining a maximum elliptical Earth orbit. The flight capsule is then released and the spacecraft propulsion is used for final orbit attainment.

A fully fueled spacecraft can also attain the Earth synchronous orbit and Mars orbit encounter trajectory.

Considerations of booster payload limitations in conjunction with test requirements led to the selection of the following three candidate test flights:

1) Test Mission A

- Planetary vehicle launch
- Nose shroud jettison at 100 nmi
- S IV B/PV injection into Earth elliptical orbit
- Capsule release and deorbit
- Spacecraft injection into escape trajectory towards Mars orbit encounter.

2) Test Mission B

- Planetary vehicle launch
- Nose shroud jettison at 100 nmi
- S IV B/PV injection into Earth elliptical orbit
- Capsule release and deorbit
- Spacecraft transfer to Earth synchronous orbit (non-Equatorial).

3) Test Mission C

- Planetary vehicle launch (with dummy capsule)
- Nose shroud jettison at 100 nmi
- Planetary vehicle injection into Earth elliptical orbit

3.4.3 Evaluation of Candidate Test Flights

3.4.3.1 Value of Candidate Test Flights

The values of the required test flight demonstrations for the three candidate test flights are shown in Table 3-2. The value of demonstration of a test flight was determined in the following manner:

- A required test demonstration was given a figure of merit of 4 if it could be met in a flight test and could not be met by a ground test.
- A required test demonstration was given a figure of merit of 3 if it could be accomplished by a flight test better than by a ground test, thus obviating the need for a ground test.

- A required test demonstration was given a figure of merit of 2 if a flight test only augments a ground test.

Table 3-2: DEMONSTRATION VALUE OF CANDIDATE TEST FLIGHTS

REQUIRED TEST FLIGHT DEMONSTRATION		CANDIDATE TEST FLIGHTS			
		VALUE OF DEMONSTRATION	A	B	C
1.	Payload Structural and Thermal Performance During Launch	3	3	3	3
2.	Payload Separation and Sequencing	4	4	4	3
3.	Capsule Deorbit & Entry	4	4	4	0
4.	Zero-g Deployments	4	4	4	4
5.	Propulsion/Attitude Control/ Structure Interaction	4	3	3	4
6.	Adequacy of Thermal Design in Space	4	4	3	3
7.	Long Life	4	4	4	4
8.	Range Transponder Design Adequacy	3	3	0	1
9.	Mission Operations	2	2	2	0
10.	Propulsion Subsystem Performance	2	2	2	2
11.	Canopus Star Tracker Performance	2	2	1	1
12.	Science Instruments Calibration	2	1	2	2
Totals		38	36	32	27

The value of the required test flight demonstration was apportioned to each candidate test mission according to the degree of capability of the test flight to satisfy that requirement. For example, the interaction of the propulsion subsystem and spacecraft dynamics and control can be evaluated better with the full inertial effects of a planetary vehicle than with just a spacecraft. Flight C with a planetary vehicle is therefore of more value than Flights A and B, with only a spacecraft, in satisfying this test requirement.

The evaluation results indicate that Test Flight A, which involves a deep space flight, is of higher value than either Test Flights B or C. This is true because Test Flight A more closely duplicates the 1973 mission.

Specifically, Test Flight A allows for the following:

- 1) Use of the DSN
- 2) Realistic thermal performance
- 3) Realistic performance of Canopus star tracker
- 4) Realistic relay link performance
- 5) Realistic deep space environment.

3.4.3.2 Impact of Candidate Test Flights or Ground Testing and Facilities

The impact of the required flight test demonstrations on ground testing and facilities is summarized in Table 3-3. A description of ground tests that may be eliminated or reduced in the event of a flight test is given below.

- 1) Structural Model Tests -- Wind tunnel model tests define the fluctuating pressure distribution and the phased relationship of the acoustic field and structural dynamic response to the shroud during launch. Full scale model tests are then performed in two steps: For the transonic regime a phased-horn array is used to provide the acoustic pressure field to be imposed on a shrouded test model of the planetary vehicle. For the liftoff environment, this test model is exposed to an F-1 engine firing at the MSFC test facility. A test flight would eliminate wind tunnel and phased horn array tests. Even though the launch vehicle is a Saturn IB and not a Saturn V, the test would verify the analyses or provide information for modifying the analytical technique.
- 2) Separation Tests -- The separation of nose cone, shroud, planetary vehicles, and capsules can be accomplished to a limited extent on the ground. A large chamber and complex supporting equipment would be required. The flight test would better simulate the actual sequence and the ground test could be omitted.
- 3) Deployment Tests -- The demonstration of deployment is partially simulated using Earth gravity compensatory devices. The requirement for such equipment and for the ground tests themselves could be replaced by a test flight.
- 4) Space Chamber Propulsion Interaction Tests -- A ground test of the interaction between the structure, the autopilot, and the propulsion subsystem requires a complex vacuum chamber test. A constraining mechanism, gimbal mechanism, and exhaust diffuser are some of the complications. This test can be better accomplished in a test flight which would replace the ground test.
- 5) Thermal-Vacuum Tests -- Following ground qualification tests, flight testing will reduce the need for thermal vacuum testing on each flight spacecraft.

Table 3-3: IMPACT OF REQUIRED TEST FLIGHT DEMONSTRATIONS

REQUIRED TEST FLIGHT DEMONSTRATION	IMPACT ON GROUND TESTING	IMPACT ON FACILITY EXPENDITURES
1. Payload Structural And Thermal Performance During Launch	Replaces Phased Horn Array Acoustic Test	Deletes Requirement for Phased Horn Array Facility
2. Payload Separation and Sequencing	Adds Confidence	None
3. Capsule Deorbit and Entry	Adds Confidence	Possibly: Those Required For Capsule Development Flight Test
4. Zero-g Deployments	Adds Confidence	None
5. Propulsion/Attitude Control/Structure Interaction	Delete Attempt at S/C Chamber Firing Test	Decreased Impact on Use of Space Chamber
6. Adequacy of Thermal Design in Space	Delete Flight S/C Acceptance Thermal Vacuum Tests	Decreases Impact on Use of Space Chamber
7. Long Life	Delete Mission Confidence Tests	Decreases Impact on Use of Space Chamber
8. Range Transponder Design Adequacy	Adds Confidence	None
9. Mission Operations	Adds Confidence	None if Flight is to Deep Space Additional Requirements, Perhaps, For Earth Orbits
10. Propulsion Subsystem Performance	Adds Confidence	None
11. Canopus Star Tracker Performance	Adds Confidence	None
12. Science Instrument Calibration	Adds Confidence	None

- 6) Mission Confidence Tests -- A mission confidence test is an extended space chamber test of a spacecraft at mission conditions (environment and functional sequences) for the purposes of developing confidence in reliability and life system components. The test flight replaces the need for such a test.

The relative effects on ground tests for each of the three candidate test flights are shown in Table 3-4. The ground tests affected, the purpose served by the ground tests, and a measure of the degree to which a flight test supplements or complements the ground test are included. If the flight test replaces the ground test, it was awarded a figure of merit equal to 100. If it only complements the ground test, a lower figure was assigned. The figure-of-merit allocation is apportioned equally to each purpose served by the test.

This evaluation shows that the three candidate test flights are of equal value in deleting and reducing ground tests and facilities.

3.4.3.3 Effect of Candidate Test Flights on Confidence

The relative effect of each candidate test flight on confidence is shown in Table 3-5. A test flight that provides opportunity for identical simulation of the 1973 mission environment and functions was given a figure of merit equal to 100. A lower value was assigned for deviations from the 1973 mission environment and function.

The confidence comparison indicates that the deep space Flight A provides a higher level of confidence than either Flights B or C. This is not unexpected since the flight was designed to simulate a Mars mission as closely as possible. Any one of the three flights provide more confidence than would otherwise be attainable from a ground test program.

3.4.3.4 Effect of Candidate Test Flights on Cost

A test flight cost summary is tabulated below:

Test Flight Cost Summary	
	<u>Cost in Millions</u>
Spacecraft	\$ 16
Experiments	25
Capsule	16
Launch Vehicle and Launch Operations	48
	<u>\$105</u>

In accordance with the guidelines for the study, the cost of mission operations are not included. On this basis, a test flight would cost approximately 105 million dollars. A test flight that includes a dummy capsule (mass and thermal properties simulated), such as Test Flight C, would reduce this figure by approximately 12 million dollars. Savings from reduced ground testing are approximately 2.2 million dollars as tabulated on the top of page 3-13.

Table 3-4: RELATIVE EFFECT OF FLIGHT TESTS ON GROUND TESTING

AFFECTED GROUND TEST	PURPOSE SERVED BY TEST	DEGREE TO WHICH TEST FLIGHT SUPPLEMENTS OR COMPLEMENTS GROUND TEST		
		CANDIDATE TEST FLIGHTS		
		A	B	C
Structural Model Tests:				
• Phased Horn Array Acoustic Test	• Verification of Analyses of Structural Response to Fluctuating Pressure Distribution in Transonic Regime	100	100	100
• F-1 Engine Test	• Verification of Structural Integrity for Launch Acoustic Conditions	50	50	50
Separation Tests	• Verification of Adequacy of —Mechanical Separation —Spatial Separation	25 50	25 50	25 40
Deployment Tests	• Verification of Adequacy of —Mechanical Separation —Dynamic Separation	25 50	25 50	25 50
Propulsion Interaction Tests	• Verification of Analyses of Autopilot Response to Interaction of Propulsion S/S & Structural Dynamics	85	85	100
Thermal Vacuum Tests	• Verification of Spacecraft Thermal Design Adequacy	100	95	95
Mission Confidence Test (Space Simulation)	• Verification of Design Adequacy	<u>100</u>	<u>95</u>	<u>95</u>
	Total	585	575	580

TABLE 3-5: RELATIVE EFFECT OF GROUND TEST AND FLIGHT TEST ON CONFIDENCE

TEST REQUIREMENT	DEGREE TO WHICH GROUND TEST SIMULATES 1973 MISSION	DEGREE TO WHICH FLIGHT TEST SIMULATES 1973 MISSION		
		FLIGHT A	FLIGHT B	FLIGHT C
1. Payload Structural & Thermal Performance in Launch Environment	80	80	80	80
2. Payload Separation and Sequence	20	100	100	80
3. Capsule Deorbit and Entry	10	90	90	0
4. Zero-g Deployments	40	100	100	100
5. Propulsion / Attitude Control / Structure Interaction	75	85	85	50
6. Adequacy of Thermal Design in Space	85	100	95	95
7. Long Life	50	100	95	95
8. Range Transponder Design Adequacy	80	100	85	90
9. Mission Operations	80	100	100	50
10. Propulsion Subsystem Performance	80	100	100	100
11. Canopus Star Tracker Performance	80	100	90	90
12. Science Instruments Calibration	80	100	100	100
	760	1155	1120	930

Potential Savings From A
Ground Test Program

Structural Tests	\$ 690,000
Separation Tests	100,000
Propulsion Interaction Tests	484,000
Flight S/C Thermal Vacuum Tests	366,000
Mission Confidence Tests	643,000
	<u>\$2,173,000</u>

Further savings could be attained by using a ground test planetary vehicle for the flight test. The cost of a ground test planetary vehicle is tabulated below.

Ground Test Planetary Vehicle Cost

	<u>Cost in Millions</u>
Ground Test Spacecraft	\$16
Ground Test Experiments	25
Ground Test Capsule	16
	<u>\$57.2</u>

The schedule permits the use of a ground test planetary vehicle for the flight test. Hence, the cost of a flight test using a ground test planetary vehicle would be approximately 46 million dollars (launch vehicle and launch operations at 48 million minus ground test program savings at 2 million).

3.4.3.5 Effect of Candidate Test Flight on Program Schedule

Significant program milestones with and without a test flight are shown in Figure 3-2.

The following guidelines were used in modifying the baseline schedule to include a test flight:

- 1) The test flight was scheduled following spacecraft qualification tests to ensure a good probability of success.
- 2) It was scheduled early enough to get nearly a year's space testing before delivery of 1973 flight spacecraft to KSC.
- 3) The 3-month standby at KSC for the baseline schedule is not required for the test flight. The 4-month average processing time for the baseline schedule at KSC was increased to 6 months for the test flight. This provides sufficient time for achieving compatibility of the spacecraft with the DSN through the DSIF 71 at KSC.
- 4) The mission confidence testing in the baseline schedule has been deleted as redundant to the test flight. The KSC checkout and DSN compatibility periods have also been deleted because the test flight spacecraft will have provided for these functions in support of the 1973 mission (assuming candidate Flights A or B are selected).

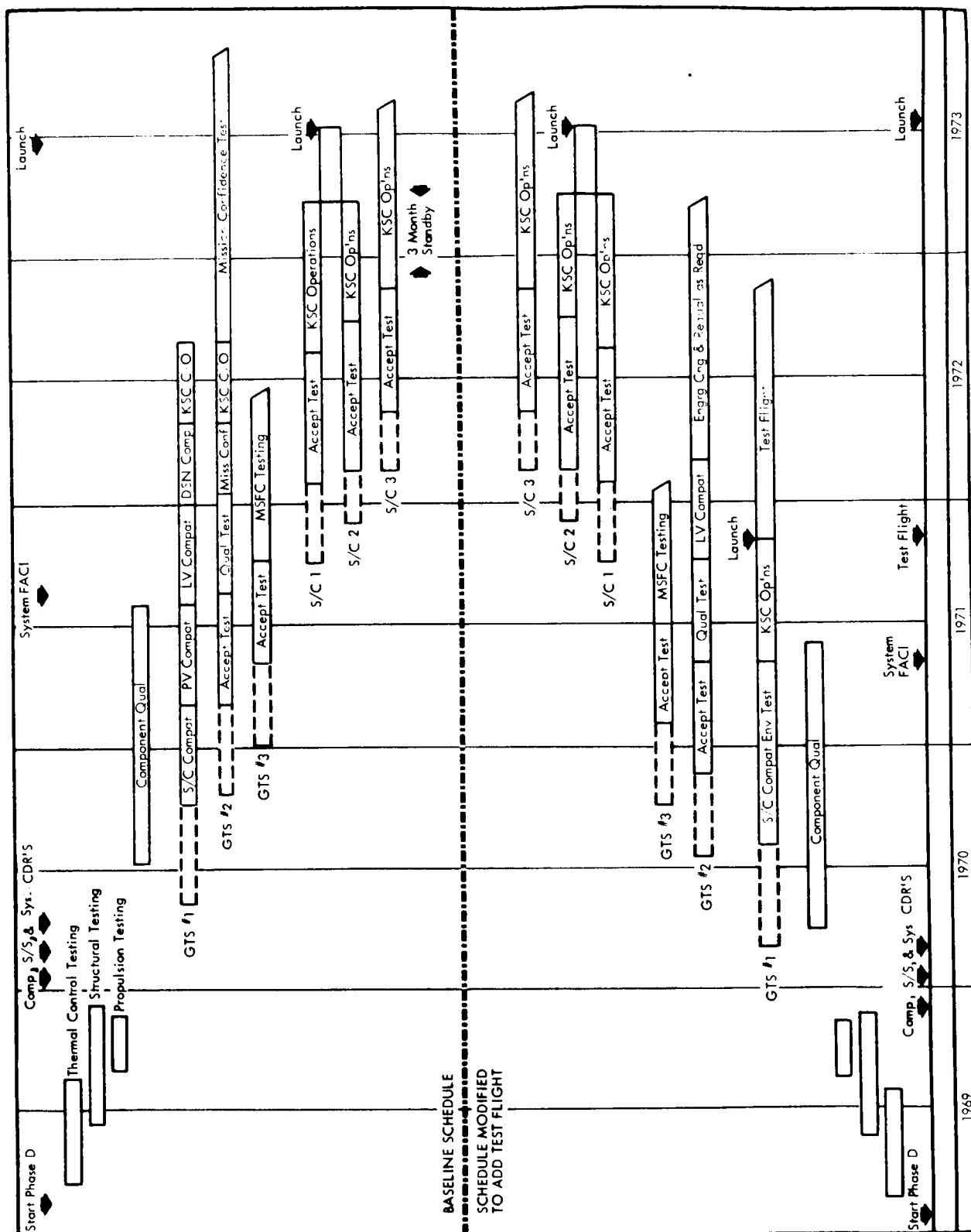


Figure 3-2: COMPARISON OF PROGRAM SCHEDULE WITH AND WITHOUT TEST FLIGHT

- 5) To support the test flight, the first ground test spacecraft schedule has been moved forward 2 months and the subsequent ground test spacecraft schedule $3\frac{1}{2}$ months. This pushes the CDR earlier by 1 month; FACI occurs $3\frac{1}{2}$ months earlier. Because spacecraft qualification occurs earlier, component qualification has been moved forward too.

The consequence in manpower loading schedule is shown in Figure 3-3.

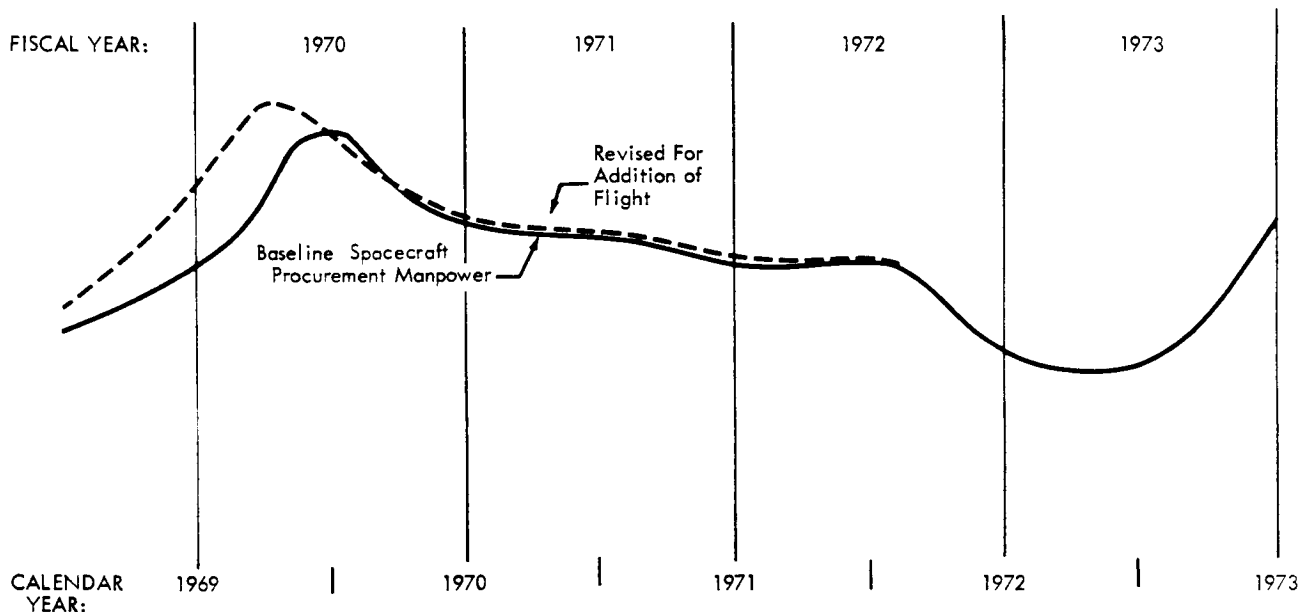


Figure 3-3: EFFECT OF TEST FLIGHT ON MANPOWER LOADING

The decrease in testing in 1972 and the increase in activity in 1970-1971 has accentuated the manpower peak. The test flight program schedule also imposes earlier and more concentrated requirements on suppliers for components to support fabrication of test spacecraft.

3.5 CONCLUSIONS

The following are the conclusions of the Voyager program test flight study:

- 1) A test flight results in a higher level of confidence for the Voyager program.
- 2) The attendant reduction of ground testing and facilities is not significant.
- 3) The most valuable test flight using the Saturn IB places the planetary vehicle in Earth orbit and the flight spacecraft on a deep space flight that encounters the orbit of Mars.
- 4) The test flight can support a 1973 mission.

- 5) Rescheduling is required to make the test flight compatible with the baseline 1973 mission schedule.
- 6) A Voyager program test flight increases program cost by approximately 50 to 100 million dollars.

4.0 SCIENCE PAYLOAD EVOLUTION

4.1 OBJECTIVES

The three primary objectives of this study task were to:

- 1) Determine the science payload evolution from the first Voyager Mars mission in 1973 to the subsequent missions in 1975, 1977, and 1979.
- 2) Develop the physical and functional characteristics of the spacecraft experiment payloads for each launch opportunity.
- 3) Determine the impact of the payload evolution on the spacecraft.

Secondary objectives were considerations of the science data automation equipment (DAE) centralization, and investigation of computer simulation for science evaluation.

4.2 APPROACH

The study approach for this task was to:

- 1) Identify the probable science objectives for the Voyager-Mars spacecraft from 1973 through 1979.
- 2) Define a hypothetical 1973 science payload baseline and describe its characteristics.
- 3) Identify potential science experiment evolution for 1975, 1977, and 1979.
- 4) Develop the physical and functional characteristics of the science experiments for the four missions.
- 5) Evaluate the impact of the experiment evolution on the spacecraft.
- 6) Identify any significant changes that should be made to the 1973 spacecraft design as a result of this impact.

4.3 RESULTS

The results of this study task are discussed below. The discussion starts with the derivation of science objectives, measurements methods, and candidate experiments leading to a hypothetical 1973 science payload. Evolution of experiments is next discussed followed by a description of the experiments' characteristics. Finally, the impact of the science payload and the significant changes to the spacecraft are discussed.

4.3.1 Science Objectives

The overall objective of space science exploration is to increase man's knowledge and understanding of:

- 1) The origin and evolution of the universe,
- 2) The origin and evolution of the solar system,
- 3) The origin and evolution of life.

This overall objective is achieved by formulating theories or hypotheses consistent with known facts and developing mathematical models to verify quantitatively that the theory is consistent with the observed facts.

These general theories will have implications leading to a hierarchy of more specific questions which can ultimately be answered by specific experiments. The information from each specific experiment will in turn result in verification, refutation, or modification of the theory or models. These in turn will lead to further questions and further experiments. This rationale is illustrated in Figure 4-1.

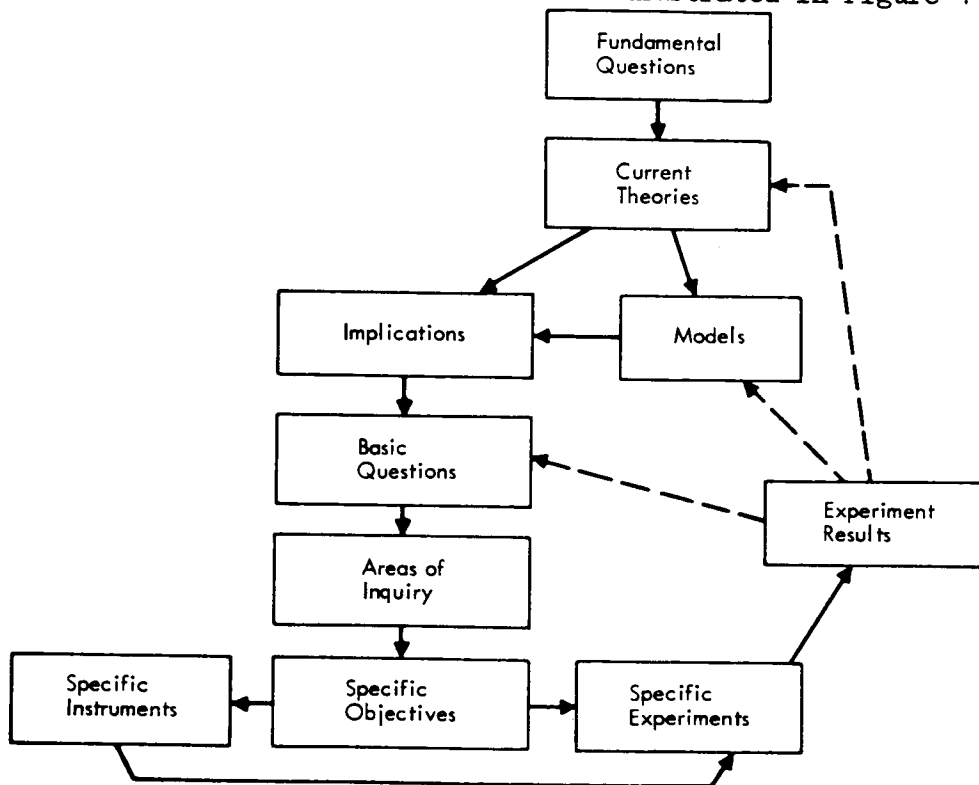


Figure 4-1: RATIONALE FOR SPECIFIC SPACE EXPERIMENTS

The Space Science Board of the National Academy of Science has outlined the overall objectives of Mars exploration in its Woods Hole Report. These objectives are the determination of (1) the origin and evolution of Mars (and the solar system) and (2) the origin and evolution of life on Mars. These objectives led to the identification of six interrelated areas of inquiry:

- 1) Composition -- This area of inquiry concerns chemical, physical, and mineral structure of the planet's atmosphere, crust, and, to the extent that can be determined, its interior.
- 2) History -- This area of inquiry concerns the path of the planet's evolution, particularly in regard to planetary size and distance from the sun, in response

to the evolutionary driving forces. Significant information about the origin and evolution of the solar system may be obtained by comparing the evolutionary paths of Earth and Mars, and verifying that Mars and Earth were formed at about the same time from similar protomaterials.

- 3) Exobiology -- This area of inquiry is concerned with the characteristics and evolutionary path of Martian life forms if such exist. If life does not exist on Mars, it is also concerned with the reasons why life did not develop. Of particular interest would be any evidence of the development of protolife forms such as organic compounds, amino acids, and protein molecules.
- 4) Differentiation -- This area of inquiry is closely associated with the history of the planet. However, the extent to which Mars has undergone a differentiation into a crust, mantle, and core has such profound significance in understanding Mars' evolution that it has been set aside as a specific area of inquiry. The basic question is whether Mars already has passed through an outgassing stage and has lost most of its atmosphere and water, or whether Mars has not gone through any extensive outgassing. The answer to these questions is important in interpreting findings in the areas of exobiology and composition.
- 5) Activity -- This area of inquiry includes the search for the major constructive and destructive forces that have been shaping or modifying the planet.
- 6) Atmospheric Dynamics -- Atmospheric dynamics, although an element in the "activity" area, has been included as a separate area of inquiry. It is of particular importance because (1) it may be one of the main destructive forces involved in the geology of Mars (areology), thus related to activity and history; (2) lower atmospheric dynamics, particularly water transport mechanisms, may have an important bearing on the location of life; and (3) a study of upper atmospheric dynamics may shed light on atmospheric loss mechanisms which relate to the evolutionary history of the planet.

The relationship of these six areas of inquiry to the overall planetary exploration objectives is indicated in Figure 4-2. Because the six areas of inquiry are closely related to each other, a single experiment will often have objectives that, when achieved, will result in answers to questions in more than one area.

For each area of inquiry there are four physical realms, or environments, of Mars that should be investigated, namely:

- o Atmosphere
- o Crust (Surface and Near Surface)
- o Interior
- o Biosphere

The four physical realms have similar experiment objectives: The determination of chemistry, structure, processes, and vestiges (traces).

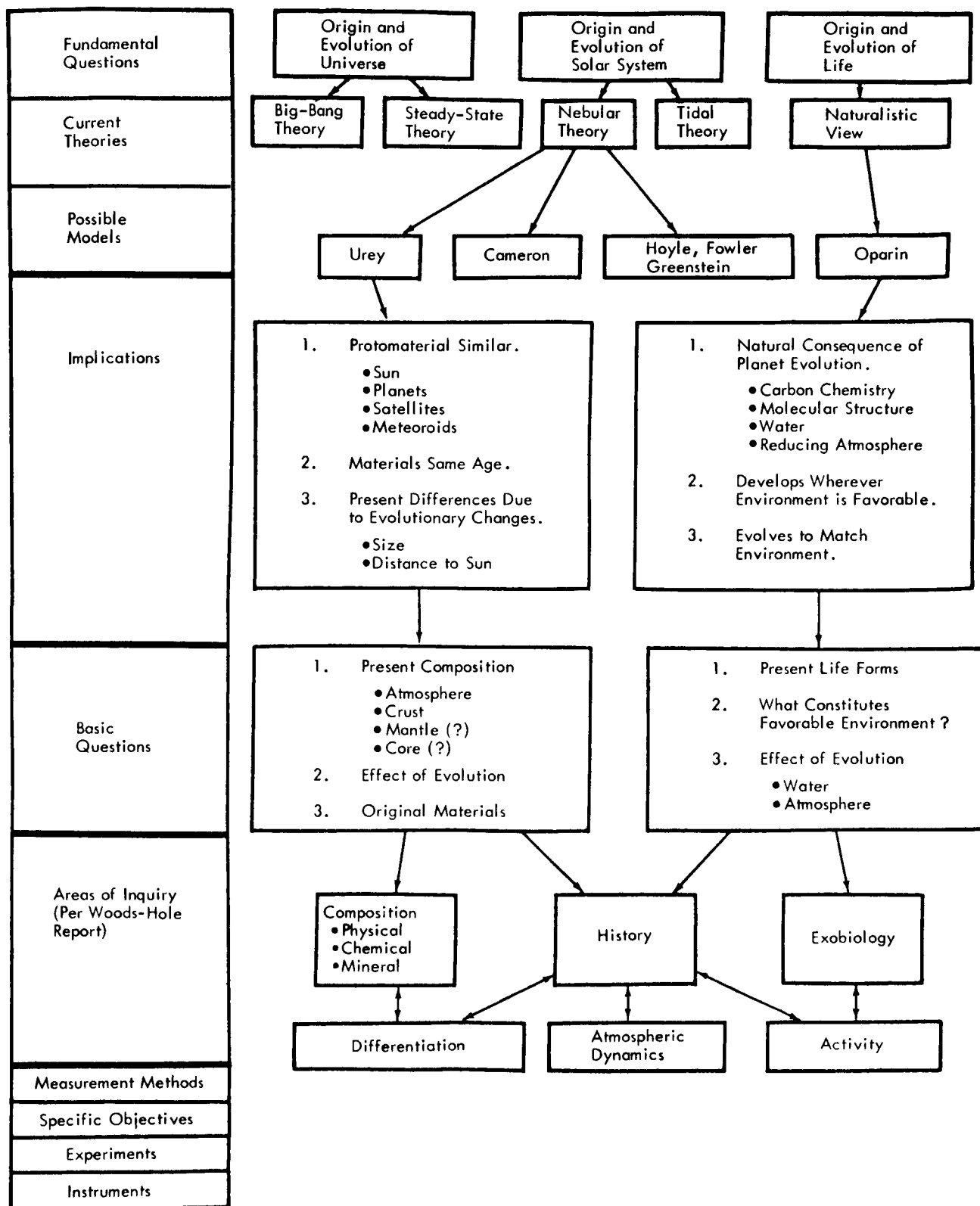


Figure 4-2: AREAS OF INQUIRY FOR PLANETARY EXPLORATION

For the first objective, major chemical constituents of the four realms are important. The second objective, determining the structure and physical characteristics of the realms, requires, for the atmosphere, measurement of the temperature or pressure profiles, storm systems, etc. For the crust, measurement relating to mountain ranges, craters, etc., are needed. In the realm of exobiology, measurements would include the actual morphology of bioorganisms.

The two foregoing experiment objectives -- chemistry and structure -- would suffice for the study of Mars as it now exists. However, a study of the evolution and history of Mars is required. This is achieved by the remaining two objectives relating to processes and vestiges (traces).

Processes are those chemical and physical processes that are currently changing Mars, and that can be measured as they occur. Studies of upper atmosphere phenomena fall within this category. The vestige measurement category requires observation of the results of earlier evolutionary processes. The measurement of trace elements such as the noble gases is an example of this class of measurements. In many cases the observations of vestiges of past evolutionary processes cannot be planned a priori. Nevertheless, the possibility of such observations is an important factor in determining the value of an experiment.

The possibility of making important measurements relative to the origin and evolution of the universe, although not a primary Voyager objective, should not be overlooked. Such measurements would be restricted to the interplanetary environment structures and processes in the vicinity of Mars and in cis-Martian space.

Candidate experiments selected for the Voyager payload should satisfy the measurement objectives developed above.

4.3.2 Measurement Methods

In selecting Voyager science payloads, it is desirable to choose experiments that use a wide variety of measurement methods. Also spacecraft candidate experiments should be selected to ensure the effective use of the orbital capability.

As indicated in Figure 4-3, the orbiter is best suited to synoptic or reconnaissance measurements, whereas the landed capsule is best used for obtaining measurements for a limited surface area. The efficient exploration of Mars requires both types of measurements.

The orbiter is used to advantage when assisting in the selection of capsule landing sites and determining how typical these lander sites are by comparing orbiter measurements of lander sites with other similar locations on Mars.

Another factor to be considered in selecting spacecraft experiments is that the orbiter will have limited opportunity to make direct measurements of Mars' characteristics. Most measurements will be remote. As indicated in Figure 4-4, remote measurements consist mainly of measuring characteristics of electromagnetic radiation either emanated or reflected from the planet.

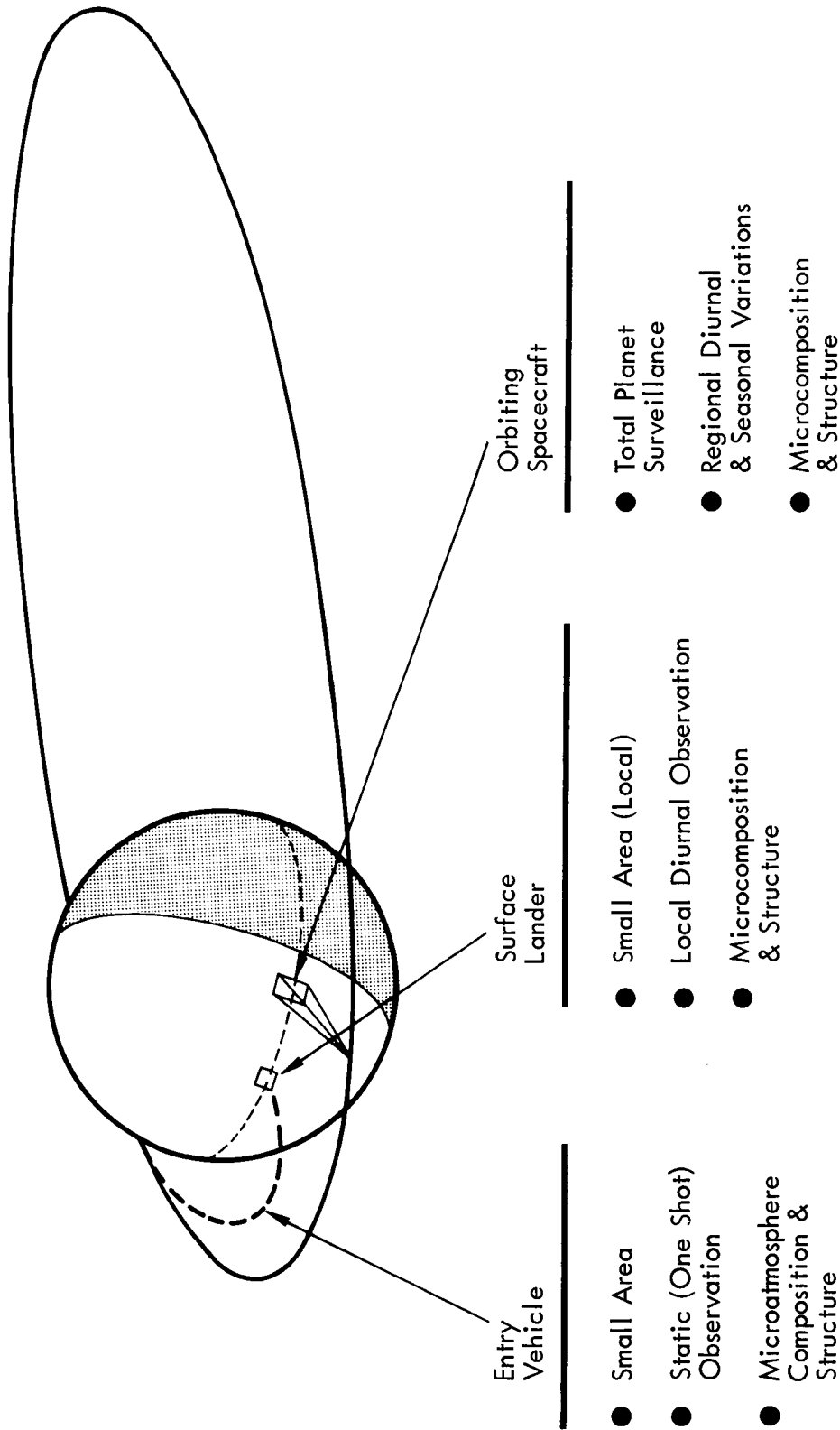


Figure 4-3: MEASUREMENT CAPABILITY OF VOYAGER PLANETARY VEHICLE ELEMENTS

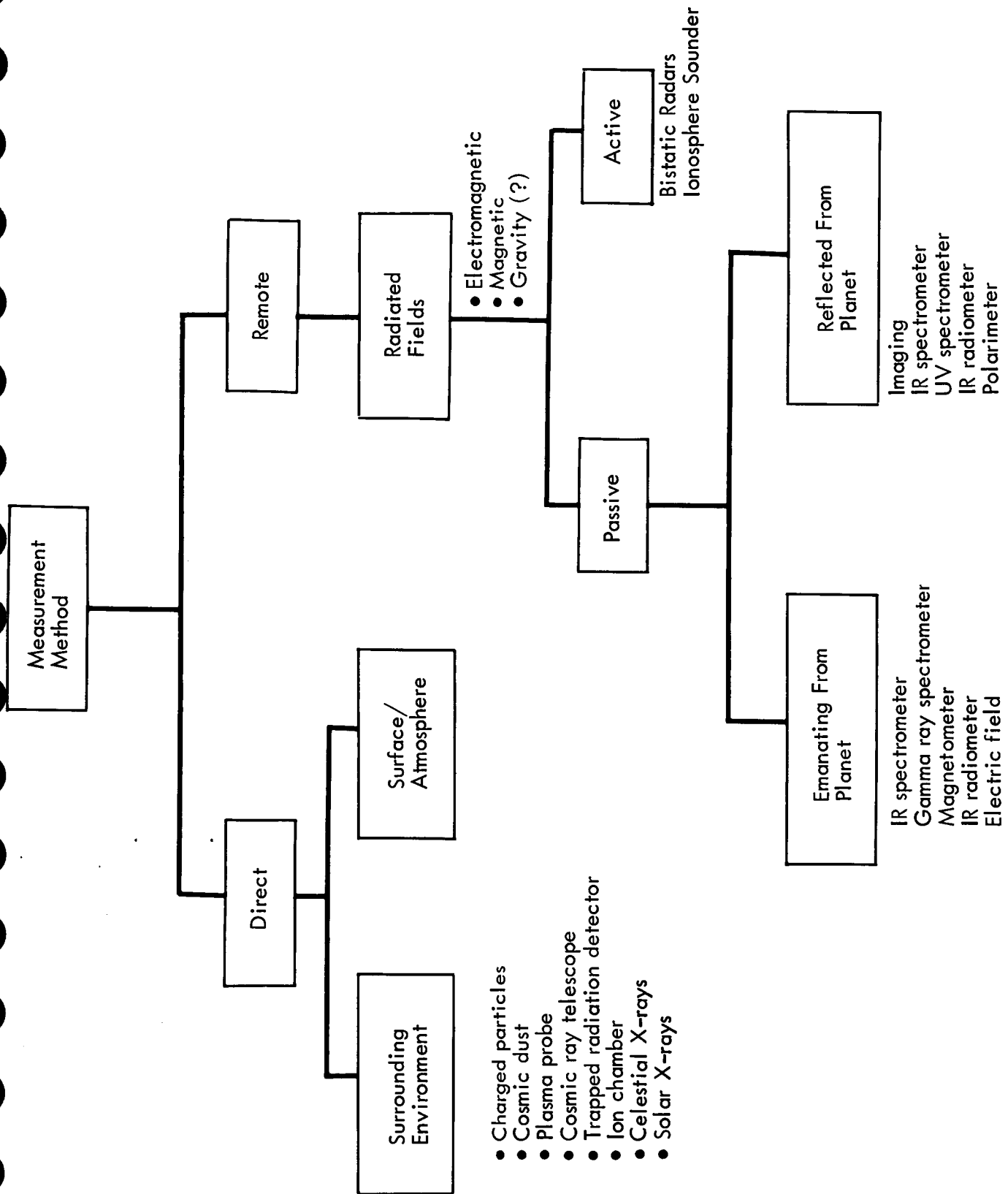


Figure 4-4: SPACECRAFT SCIENCE EXPERIMENTS MEASUREMENT METHODS

The regions of the electromagnetic spectrum of particular interest to the spacecraft science payload are: (1) infrared (IR), (2) visible, and (3) ultraviolet (UV). Measurements methods selected for observing the electromagnetic spectrum of interest depend on the electromagnetic environment of Mars.

The reflected solar spectrum and the radiation spectrum emitted from the planet are shown in Figure 4-5. Also shown in the figure is the absorption spectrum. The data show that below 1900 angstroms, a broad CO₂ absorption continuum will prevent solar radiation from reaching the surface. Therefore, at the shorter wavelengths only upper atmosphere phenomena can be observed. At wavelengths below 1 angstrom (not shown), the atmosphere will become transparent again and this region will be useful for the gamma ray spectrometry.

In the regions from 2500 angstroms to about 300 angstroms (an approximate state-of-the-art limit for conventional UV spectrometers), atmospheric absorptions/reflections phenomena and fluorescence phenomena can be observed. In the regions between 2500 angstroms and several microns, the complex spectra resulting from the surface materials and the confusing effects of surface reflections will probably prevent useful spectral measurements. Consequently, only photoimaging data will be obtained in this spectral region.

At wavelengths on the order of several microns (2.5 to 30), molecular vibration spectra predominate. Measurements in this region may give information as to the presence of certain molecules. Organic compounds would be of particular interest, although their spectra would probably be too complex for complete identification. However, concentration of organic compounds might be inferred, thereby facilitating the location of candidate landing sites. In this region (2.5 to 30 microns), advantage can be taken of the several CO₂ absorption bands to establish atmospheric temperature profiles. Spectral measurements in the far infrared (beyond 50 μ) will probably not yield useful surface temperature data. The far infrared is the regime of pure rotational spectra. Since solids and liquids in general have no rotational bands, only atmospheric phenomena would be observable.

4.3.3 Candidate Experiments

Candidate spacecraft experiments have been selected that satisfy the Voyager mission scientific objectives. These experiments are tailored to measurements from orbit and reflect the measurement methods discussed above. The contribution of the candidate experiments to the exploration and investigation of the four physical realms of Mars -- atmosphere, crust, interior, biosphere -- are indicated in Table 4-1. The secondary contribution of these experiments to the exploration and investigation of the interplanetary environment near Mars is also indicated.

4.3.4 1973 Hypothetical Science Payload

The specific objectives of the 1973 Voyager spacecraft mission are orbital reconnaissance of Mars to characterize the planetary environment and selection of sites that possess interesting scientific characteristics. A secondary objective of the spacecraft is to perform scientific measurements during Earth-Mars transit.

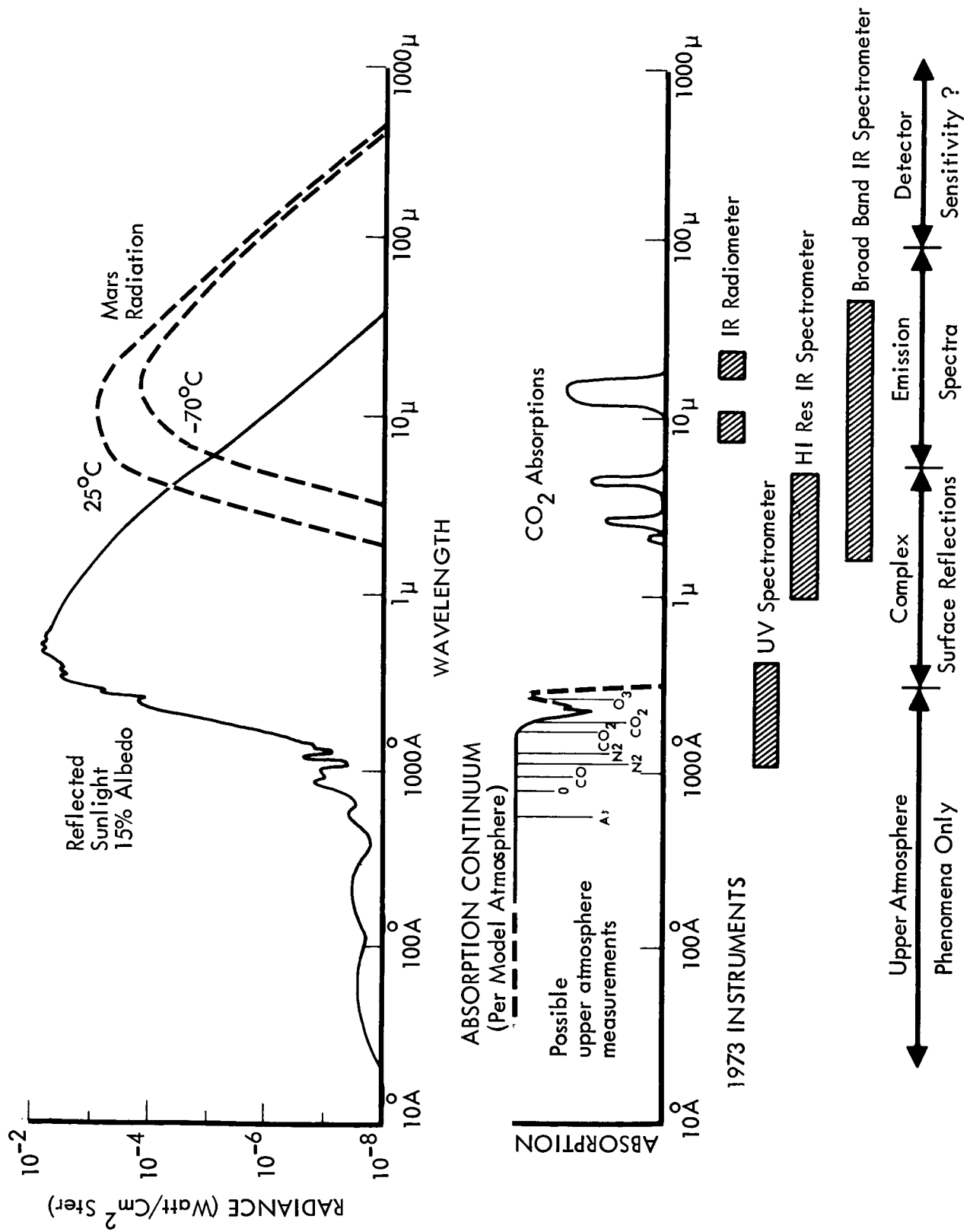



Figure 4-5: USEFUL ELECTROMAGNETIC SPECTRUM AT MARS

Table 4-1: CONTRIBUTION OF THE SELECTED SCIENCE EXPERIMENTS TO SPACE EXPLORATION OBJECTIVES

Objectives Orbital Experiments	Atmosphere				Crust				Interior				Biosphere				Mars Vicinity	
	Chemistry	Structure	Processes	Vestiges	Chemistry	Structure	Processes	Vestiges	Chemistry	Structure	Processes	Vestiges	Chemistry	Structure	Processes	Vestiges	Structure	Processes
Imaging			X			X		P				P	P					
Broadband IR Spectrometer	X				P	P	P		P				P			P		
High Resolution IR Spectrometer	X	X	X		P		P	P								P		
IR Radiometer		P	X			X	P				P				P			
UV Spectrometer	X	X	X	P														
Fields & Particles 										P	P						X	X
Atmospheric Polarimeter		X	X				P											
Atmospheric Mass Spectrometer	X		P															
Bistatic Experiment		X	X															
Bistatic Radar						P	P											
Solar Occultation	X	X	X	P														
Gamma Ray Spectrometer					X	P	P	P		P		P					X	P
Neutron Albedo		P			P												P	P
Meteor Flux							P										P	P
Cosmic Dust Detector																	P	P
Ionosphere Sounder		X	X			P				P								
Earth Occultation	P	X	X			X												
Celestial Mechanics						P		P		X		P						

X — Definite information P — May contribute, depending on nature of phenomena

 Includes magnetometer, plasma probe, trapped radiation detector, ion chamber, cosmic ray telescope and solar & celestial x-ray experiments.

The achievement of these objectives requires the following types of observations:

- 1) Extensive Mars surface imagery at medium resolution.
- 2) Limited imagery at high resolution.
- 3) Atmospheric composition, temperature, and density measurements.
- 4) Radiometric observation of Mars surface temperature.
- 5) Field and particles measurements.
- 6) Martian gravitational potential.

The specific experiments that have been selected for achievement of the 1973 spacecraft objectives are listed in Table 4-2.

Table 4-2 1973 SPACECRAFT BASELINE SCIENCE PAYLOAD

● Imaging	● Fields and Particles
● Broadband IR Spectrometer	Plasma Probe
● High Resolution IR Spectrometer	Cosmic Ray Telescope
● IR Radiometer	Trapped Radiation Detector
● UV Spectrometer	Ion Chamber
● Cosmic Dust Detector	● Atmospheric Polarimeter
	● Atmospheric Mass Spectrometer

The first five experiments will provide data on the indigenous Mars environment. The other experiments selected for the 1973 spacecraft are primarily (1) cosmic dust detection, (2) fields and particles experiments conducted during interplanetary cruise as well as in Mars orbit and (3) the atmospheric mass spectrometer and polarimeter experiments.

Two additional experiments to be conducted by the orbiter do not require on-board scientific instruments. They are the Earth occultation experiment and the celestial mechanics experiment, which use the S-band DSIF tracking signal.

In fabricating the spacecraft, care will be taken to maintain magnetic cleanliness within cost constraints. Even so, the magnetic environment of the spacecraft will be such as to exclude a magnetometer experiment unless it is mounted on an extremely long boom. This is considered impractical. Consequently, a magnetometer experiment is not proposed for the 1973 mission. A magnetometer is proposed as part of a future subsatellite experiment, which cannot be conducted in 1973 because of the 390-pound science payload weight allocation.

4.3.5 Experiment Evolution

The 1973 instruments, experiments, and their objectives will change with subsequent Mars missions in 1975, 1977, and 1979. Science experiments and support equipment can evolve in diverse ways, as shown in Figure 4-6. Experiment evolution may be achieved by using a 1973 instrument on subsequent missions to obtain extended ground and seasonal coverage. Evolution may also lead to conducting the same 1973 experiments with instruments of different spectral and spatial resolutions and spectral ranges. Finally, evolution may also lead to altogether different experiments. Experiment evolution may lead to changes in the science support equipment such as data handling techniques and instrument mounting.

Each 1973 experiment was examined to determine how it could evolve in subsequent missions. A summary of the evolutionary trends for the 1973 instruments is shown in Figure 4-7. The evolution of instruments introduced in 1975 and 1977 is also indicated.

The experiments that will contribute most to the fulfillment of the Voyager scientific objectives are photoimaging, UV and IR spectrometer, and IR radiometer. Their evolution from the 1975 to 1979 mission therefore merits detailed consideration. A study of photoimagery systems for Voyager, including evolution considerations, has been conducted and documented in Section 6.0 of this volume. Consequently, only the evolution of the UV and IR spectrometers and the IR radiometer will be discussed below.

Experiment parameters considered are spectral resolution, spatial resolution, and spectral coverage for the spectrometers; and spatial resolution, temperature resolution, and spatial coverage for the radiometer.

- 1) Ultraviolet Spectrometer and Atmospheric Sounding High Resolution (Infrared Spectrometer) Experiments -- Although the ultraviolet spectrometer and the atmospheric sounding experiments operate in different spectral regions, they both use essentially the spectrometer configuration shown in Figure 4-8.

Increased Spectral Resolution -- As long as the diffraction limit of the system is not exceeded, the spectral resolution may be increased by (1) decreasing the widths of the entrance and exit slits, (2) decreasing the grating spacing, (3) increasing the focal length of the collimating optics, and (4) operating in a higher order. Each of the four methods of increasing the spectral resolution has limitations.

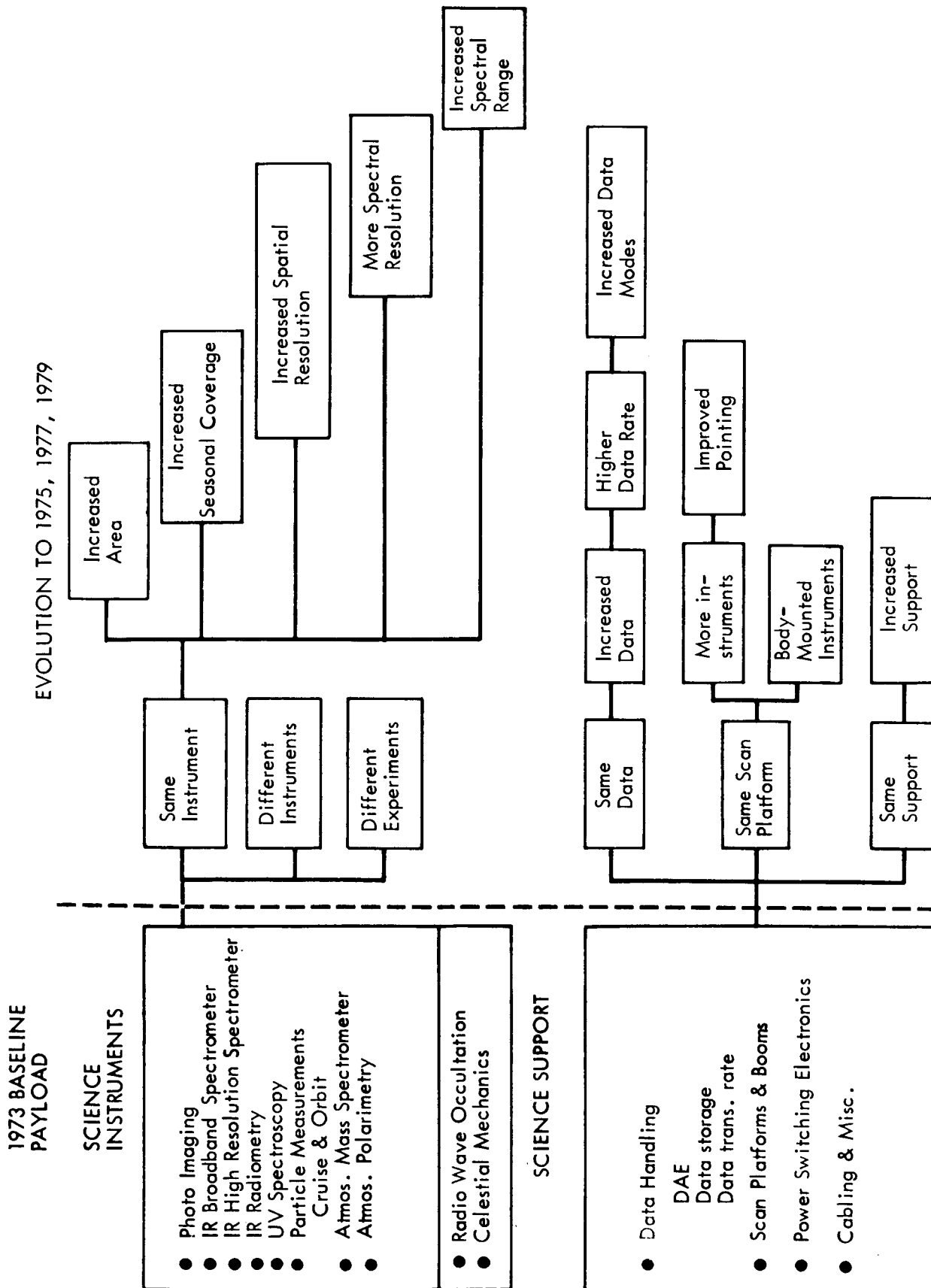


Figure 4-6: SCIENCE EXPERIMENTS EVOLUTION TRENDS

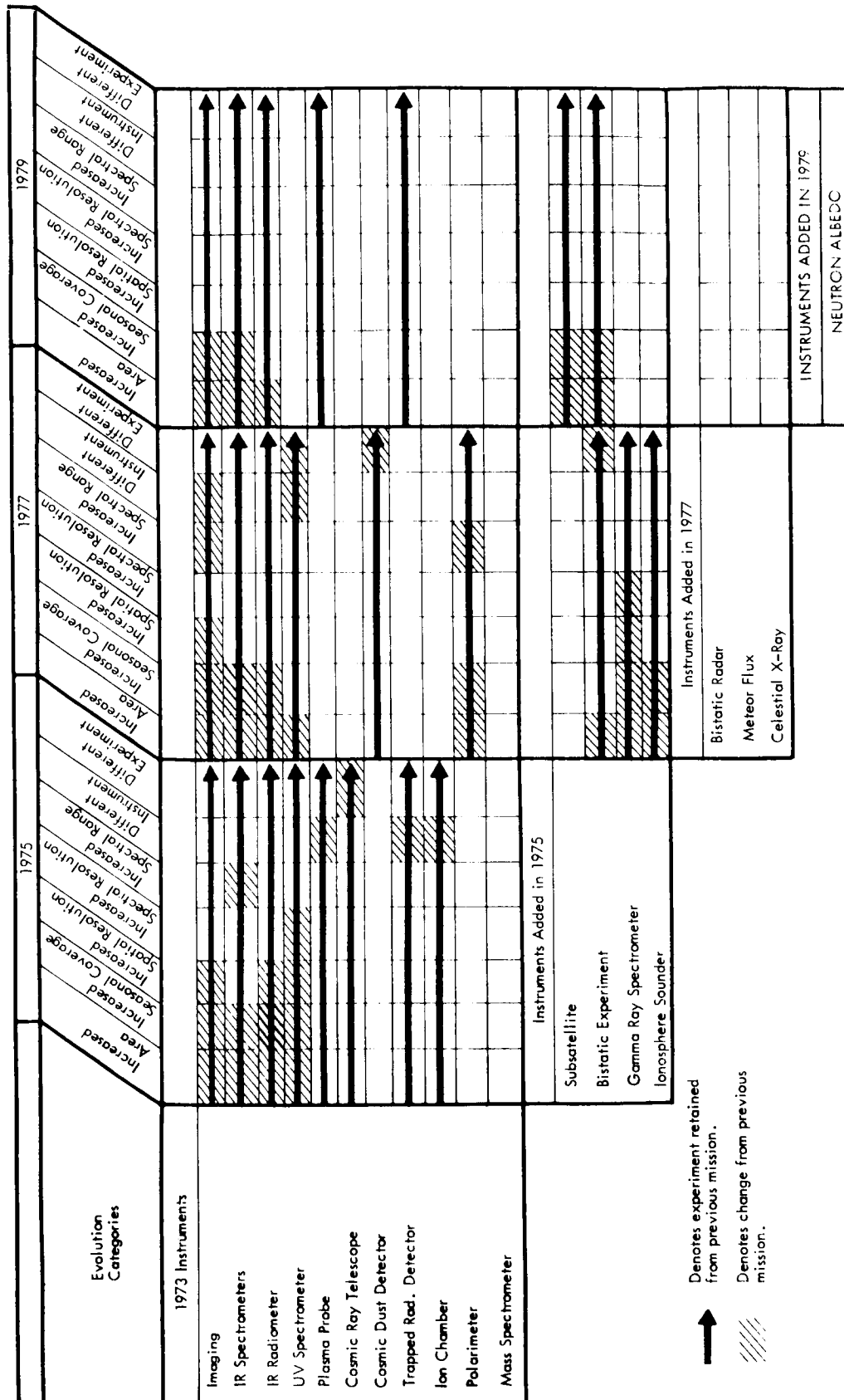


Figure 4-7: SCIENCE INSTRUMENT EVOLUTION

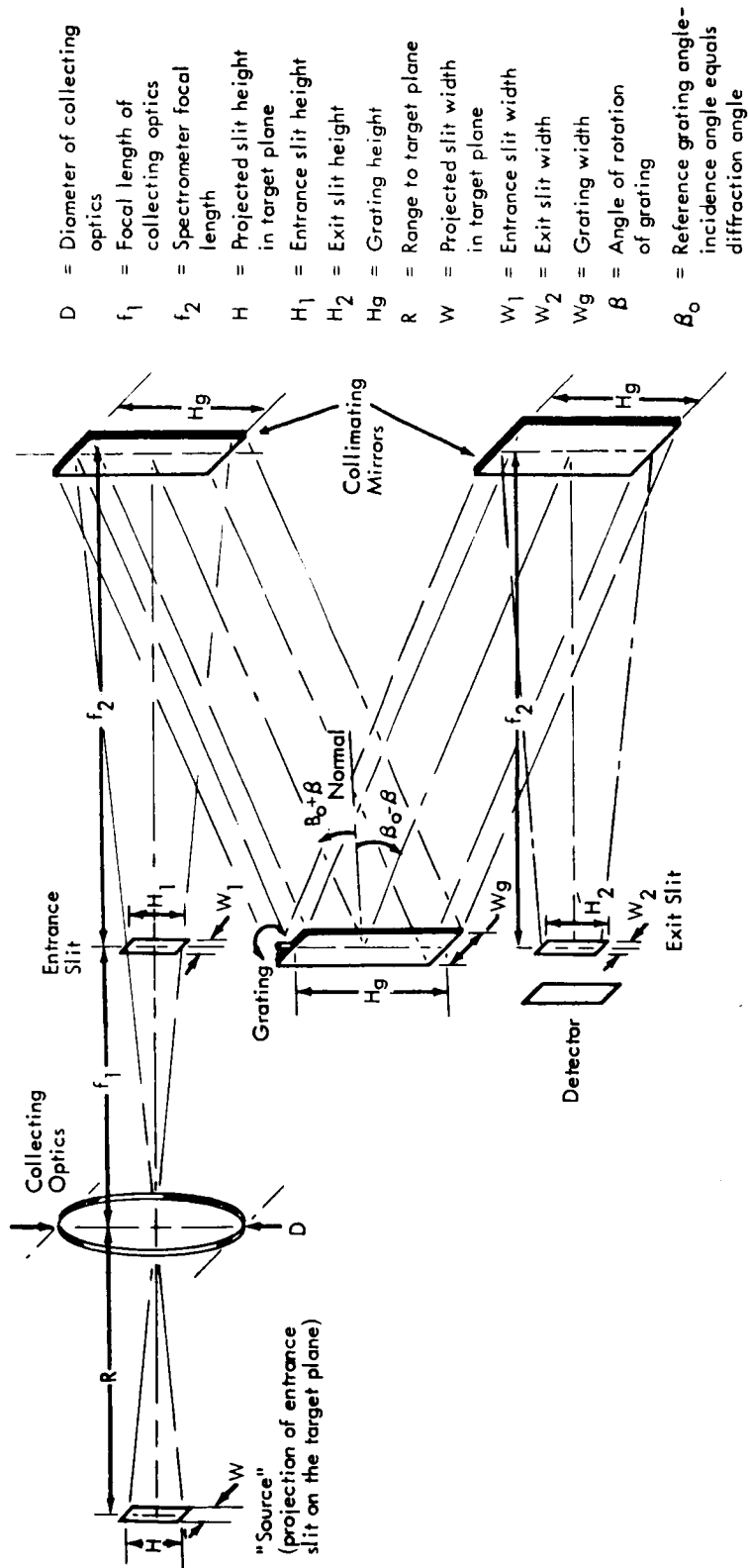


Figure 4-8: SCHEMATIC SPECTROMETER ARRANGEMENT

Operating in a higher order requires filtering to eliminate the flux from the overlapping orders. Increasing the focal length of the collimating optics requires an increase in the volume of the equipment. Decreasing the grating spacing is limited by the state of the art to about 2400 lines per mm and is costly. Decreasing the slit widths is the simplest and least costly of the methods.

If the desired spectral resolution is beyond the diffraction limit of the existing system, it is necessary either to operate in a higher spectral order, or to increase the number of grating lines. The latter can be accomplished either by decreasing the line spacing on a fixed size grating, or increasing the size of the grating with the same line spacing. Increasing the grating size requires a corresponding increase in the size of the collimating optics.

Increasing the spectral resolution generally decreases the flux incident on the detector, thereby decreasing the signal-to-noise ratio of the system. If the system was designed for operation at the original signal-to-noise ratio, either the signal flux level must be increased to the original value, or the system must be modified to operate with a lower signal flux level. The former requires one or more of the following: (1) increasing the optical efficiency of the system, (2) increasing the grating efficiency, (3) increasing the entrance and exit slit height, or (4) decreasing effective aperture ratio of the collimating optics. Increasing the slit heights is limited by aberrations in the optical system. If the initial system were designed optimally, no improvement by increased slit lengths, optics efficiencies, or grating efficiency may be expected. Increasing the effective aperture ratio of the collimating optics entails an increase in the sizes of the grating, the collimating optics, and the collecting optics. This result could have been anticipated since more flux must be introduced into the system.

The second way of restoring the required signal-to-noise ratio of the system is by modifying it to operate at the lower signal flux level. Assuming the incident flux and the detector size are fixed, only three possible changes can be made. The first is to increase the detectivity of the detector by cooling it. The second is to use a more sensitive detector. The third is to increase the time allowed to collect the data.

Estimated Spectrometer Weight Versus Spectral Resolution -- An increase in spectral resolution increases the spectrometer weight, as shown in Figure 4-9.

Increased Spatial Resolution -- The spatial resolution of the spectrometer is determined by the focal length of the collecting optics, the size of the entrance slit, and the distance from the spectrometer to the target surface. The range is dictated by the mission and is already fixed. Changing the entrance slit dimensions would effect the spectral resolution. Thus, the most reasonable change is to increase the collecting optics' focal length to decrease the ground element area determined by the entrance slit area. However, to maintain the same image radiance, the f-number of the optical system must be kept constant. Consequently, the diameter of the

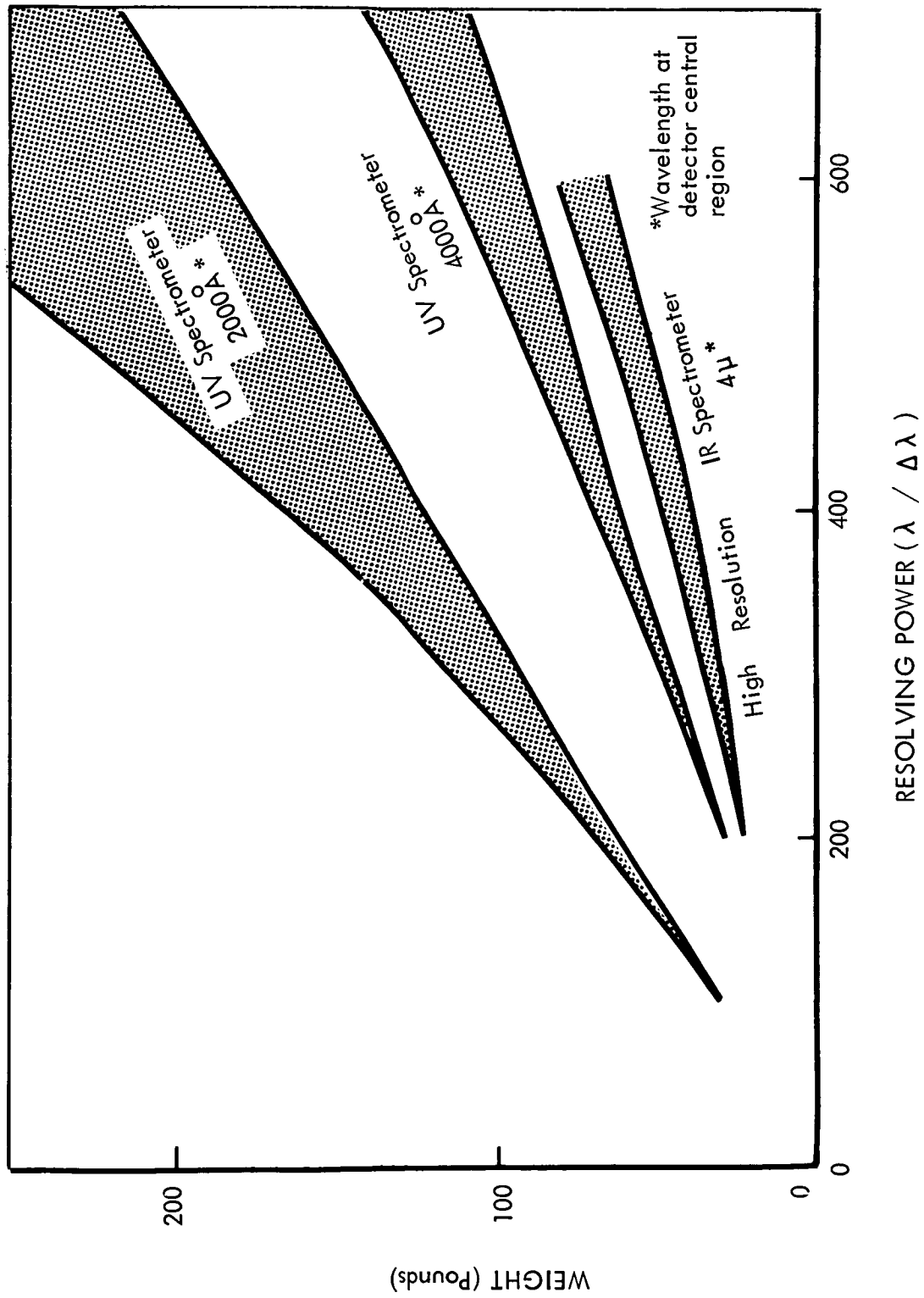


Figure 4-9: ESTIMATED SPECTROMETER WEIGHT VS. SPECTRAL RESOLUTION

optical system must be increased by the same factor as the focal length. Thus, if the ground area resolution is increased, the area of the collecting optics must also be increased by the same factor, and the optical system's weight hence will be increased by approximately the same factor, assuming lightweight mirror fabrication techniques.

Increased Spectral Range -- Techniques for increasing the spectral range of a spectrometer depend on the specific instrument's characteristics.

The simplest technique to be considered applies when the detector used in the spectrometer is sensitive over a larger spectral range than the one of interest. Since a detector has a spectral region of highest detectivity, increasing the spectral range requires the detector to operate in a region of reduced detectivity. To supply an incident flux large enough to give satisfactory operation in this region, the optics within the system must be increased. This increase is inversely proportional to the ratio of the detectivities in the two spectral regions. The optical system weight would be increased by a factor approximately equal to that of the aperture area increase.

Since increasing the spectral bandwidth increases the number of spectral elements to be scanned, either the scan time must be increased for a constant data rate, or the areas of the optical elements must be increased to allow an increased data rate.

- 2) Broadband Infrared Spectrometer -- The broadband infrared spectrometer is a collecting optical system with a portion of an interference filter wheel in its image plane. The flux transmitted by the interference filter is collected by a second optical system and imaged on a detector.

Increased Spectral Resolution -- Maintaining a fixed field of view while increasing the spectral resolution is obtained by decreasing the ratio of the entrance pupil diameter to the diameter of the filter. To maintain the same flux at the detector (so the signal-to-noise ratio will be constant), the area of the entrance pupil (collecting optics) must vary inversely with the filter bandwidth required. Thus, reducing bandwidth to $1/n$ of its value requires the entrance pupil diameter to be increased by a factor \sqrt{n} and the filter diameter to be increased by a factor of n . The change in the ratio of the collecting optics to filter wheel diameter requires an increase in the focal length of the system. The f-number of the first optical element will increase by a factor n . The secondary optical system, beyond the filter, must likewise have its focal length changed by a factor of n to maintain the same detector size.

Increasing the spectral resolution will increase the weights and costs of the primary mirror, the filter wheel, the drive motor, and the support structure. With the use of lightweight fabrication techniques, the increases in weight and cost can be assumed to vary approximately linearly with increased spectral resolution.

Increased Spatial Resolution -- The spatial resolution of the spectrometer can be increased without affecting the other parameters of the system by increasing the focal length of the primary optics while maintaining the same f-number (i.e., increasing the optics diameter also).

Increased Spectral Range -- The techniques for increasing the spectral range of the broadband IR spectrometer are similar to those for the high resolution IR spectrometer. When increasing the spectral range for this instrument, however, special attention should be given to the interference filter. This filter is on a circular plate. Increasing the spectral range requires either increasing the diameter of the plate so the "taper" of the interference films remains the same, or increasing the "taper" on the same plate. The former method increases instrument size and weight. The latter may reduce the spectral resolution.

- 3) Infrared Thermal Strip Mapping Unit (Radiometer) -- This infrared mapping unit produces a map of the thermal emittance within the scanned strip. If the spectral bandwidth of the detector is in an atmospheric window, an approximate temperature distribution of the surface is obtained. When the detector spectral band coincides with a strong atmospheric absorption band, the temperature distribution of the atmosphere is mapped.

There are three classes of radiometers. The first is the push-broom type which uses a long linear array of detectors arranged perpendicular to the velocity of the spacecraft. The velocity of the vehicle provides the scan to map a strip in the direction of motion.

The second type is the image field scanning type, which uses a mechanically driven mirror to scan across the image field. The scan generates a strip perpendicular to the vehicle's velocity. The result is a sawtooth scan of the surface.

The third type is the object field scanning radiometer. It is similar to the image field scanning type except that the scanning mirror is in the image field.

A scanning-type-radiometer is illustrated in Figure 4-10. The spatial resolution of the radiometer can be improved by (1) increasing the focal length of the collecting optics and (2) decreasing the detector size. The spatial coverage can be increased either at the expense of other system parameters, or by increasing the weight and volume of the instrument.

- 4) Cooling Requirements for Infrared Sensors -- Longer wavelength infrared sensors generally require cooling. Temperature requirements for various detectors are shown in Figure 4-11.

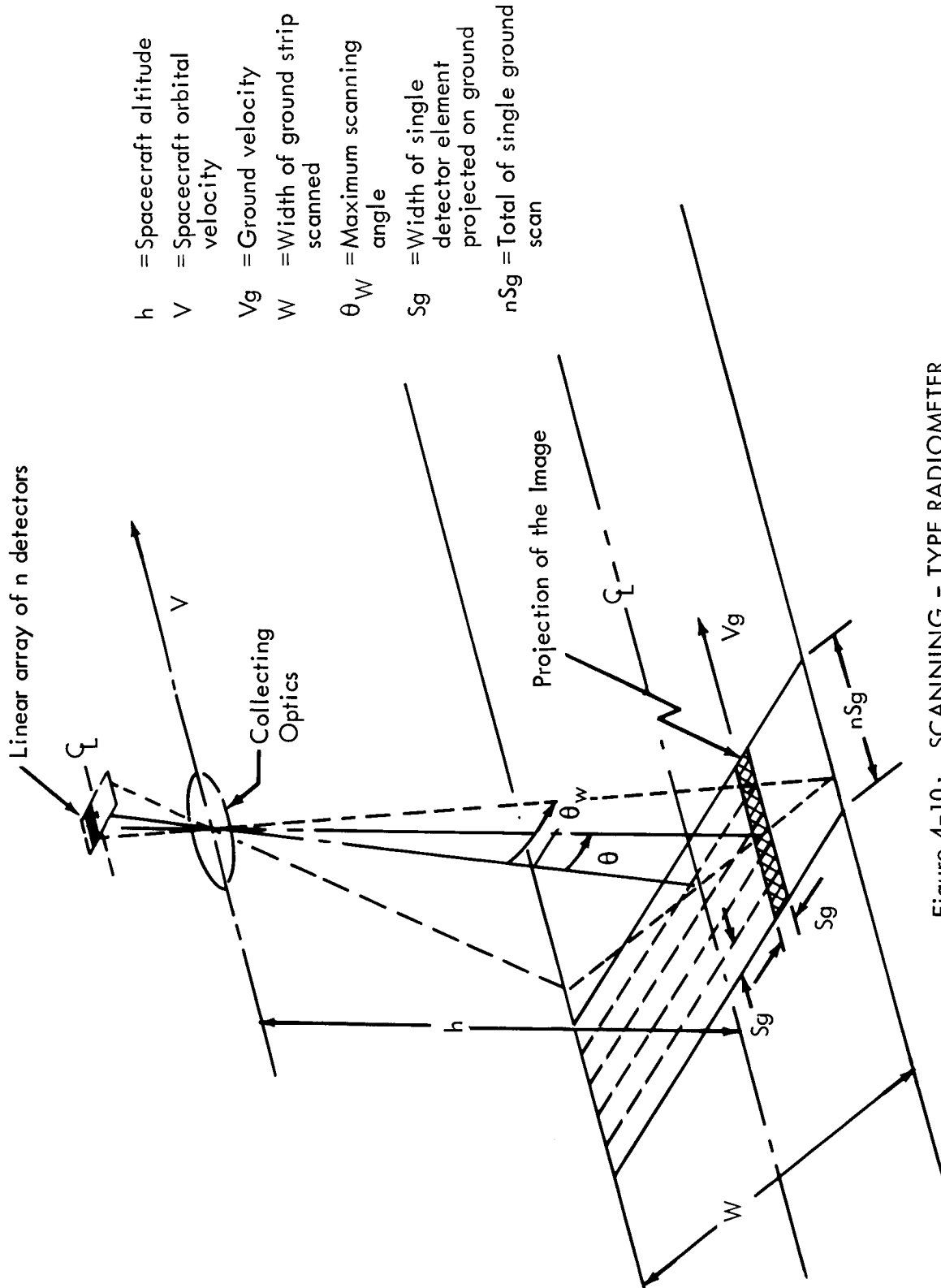


Figure 4-10: SCANNING - TYPE RADIOMETER

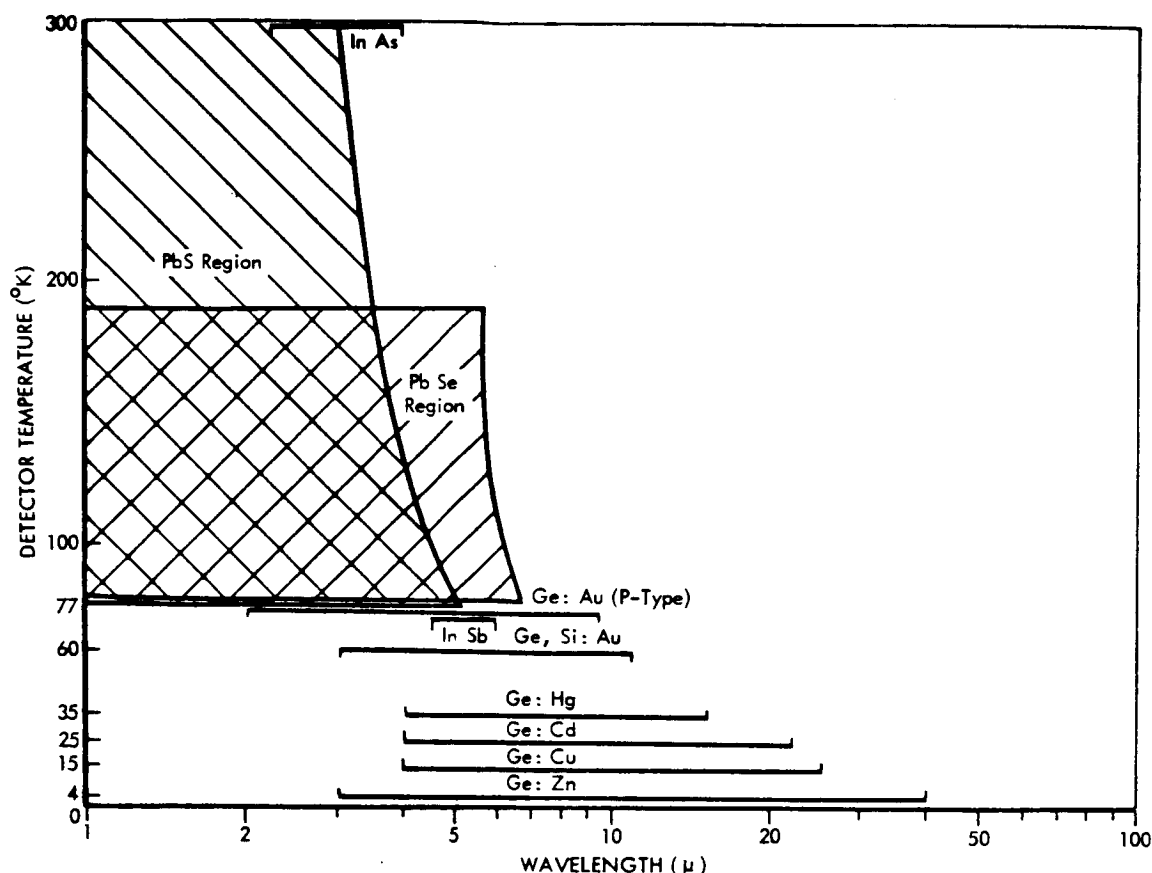


Figure 4-11: DETECTOR TEMPERATURE REQUIREMENTS AS FUNCTIONS OF SPECTRAL BANDWIDTH

In general, the longer the wavelength at which the detector operates, the greater the amount of required cooling.

The weight and power requirements for cooling the detectors vary greatly with (1) the required detector temperature, (2) the external temperature, and (3) the method of cooling. For moderate temperature requirements, thermoelectric cooling is possible. Such cooling requires modest weights and large amounts of electrical power. In space applications, radiative cooling is feasible at the expense of weight. For large temperature reductions, regenerative cooling employing expansion of gas is required, and multiple-stage cooling (i.e., nitrogen/helium) is often used. These systems may be either "open" systems employing pressurized gas in storage containers for measurements over a short period of time or "closed" systems recycling the gas for measurements over a long time period. Weight trend data required by a 1-watt closed cycle cooling system is given in Figure 4-12, as a function of sensor operating temperature. The data shows that the amount of weight required to provide sensor operating temperature below 10°K is prohibitive.

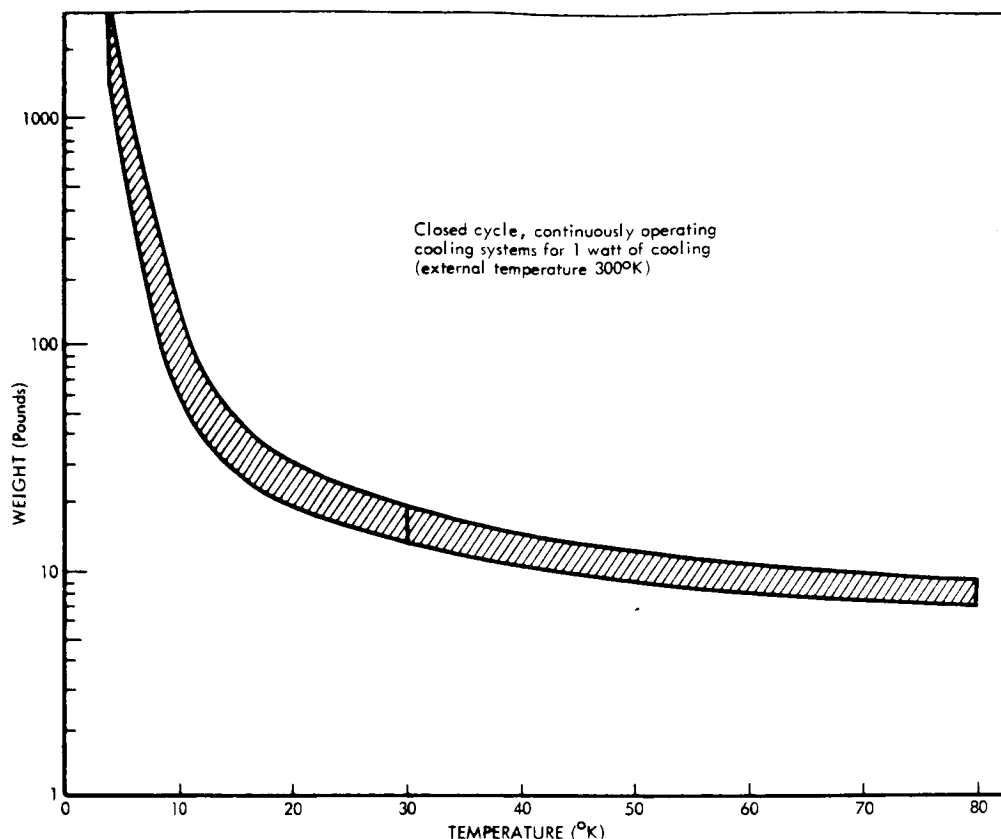


Figure 4-12: COOLING SYSTEM WEIGHT AS A FUNCTION OF IR DETECTOR TEMPERATURE

Beyond about 7 microns, cryogenic cooling (to less than 100°K) will be necessary. To take advantage of the orbiter's stay time in orbit, the use of closed cycle systems is recommended. Power requirements for such systems restrict their use to temperatures about 75°K or above. This means that general survey types of experiment should be confined to wavelengths below about 10 microns.

"One shot" experiments at larger wavelengths might be feasible using open cycle cooling; however, most of the phenomena measured would probably be more easily measured by the lander.

4.3.6 Experiment Characteristics

The functional and physical characteristics of each of the experiments considered in the science payload evolution are summarized in Tables 4-3 through 4-23. Each summary sheet contains information covering:

- Experiment objectives
- Instrument design characteristics
- Experiment requirements
- Instrument physical and functional characteristics for the applicable missions.

Table 4-3: EXPERIMENT SUMMARY SHEET

EXPERIMENT NUMBER No. 1		PHOTO IMAGING	PHYSICAL AND FUNCTIONAL CHARACTERISTICS				
EXPERIMENT OBJECTIVES:			1973	1975	1977	1979	
1973:	Photographic coverage of majority of planet (60°S to 40°N) at medium resolution (300 meters) and coverage of selected portions at higher resolution (10 meters).		150	165	660	660	
1975:	Same as 1973 but with increased planet area and seasonal coverage and with higher resolution (5 meters) of selected portions.		60	100	200	200	
1977:	Improved resolution coverage of selected portions of planet surface with dual capability of 10 and 1 meter resolution.		1.9	1.9	65	65	
1979			5 x 10 ⁵	5 x 10 ⁵	2.5 x 10 ⁷	2.5 x 10 ⁷	
			5.7 x 10 ⁸	5.7 x 10 ⁸	2.9 x 10 ¹⁰	2.9 x 10 ¹⁰	
			1.4 x 10 ¹¹	2 x 10 ¹¹	6 x 10 ¹²	6 x 10 ¹²	
			19 min	19 min	19 min	19 min	
			0.001	0.001	0.001	0.001	
			*	*	*	*	
			*	*	*	*	
			2 and 20	2 and 20	2 and 20	2 and 20	
INSTRUMENT DESIGN CHARACTERISTICS:							
1973:	Trivideon system; 2 medium resolution, 1 high resolution vidicons with electrostatic deflection.						
1975:	Same as 1973, possibly with improved optics and SEC electromagnetic vidicon for high resolution.						
1977: & 1979	Dual lens silver halide photographic system with backup vidicon.						
EXPERIMENT REQUIREMENTS:							
Operation near terminator (50° to 90° phase angle) Mount on scan platform in 1973 and 1975; mount fixed in spacecraft in 1977 and 1979.							
REMARKS:							
See Section 6.0 of this document, D2-115002-4, for further details. * Depends on particular spacecraft design and final orbit selection.							

Table 4-4: EXPERIMENT SUMMARY SHEET

EXPERIMENT NUMBER No. 2	BROADBAND INFRARED SPECTROMETER	PHYSICAL AND FUNCTIONAL CHARACTERISTICS	1973	1975	1977	1979
<u>EXPERIMENT OBJECTIVES:</u> 1973: 1) Measure quantities relevant to existence (present or past) of life a) Presence of oxidizing or reducing atmosphere b) Presence of polyatomic molecules c) Compositional variations of atmospheric constituents (geographic) 2) Collect data on surface composition, gas temperature, surface albedo, surface temperature, and atmospheric photochemistry.		Weight (pounds) Power (watts) Volume (cubic inches) *Data (bits/second) (total bits) Field of view (deg)	25 5 800 1330 for 1 orbit 665 for others 2.4x10 ⁶ for 1 orbit 1.2x10 ⁶ for other orbits 5 x 5			
<u>INSTRUMENT DESIGN CHARACTERISTICS:</u> 1973: o An f/2, 10 in dia. collector and a V-type beamsplitter to provide flux to two circular interference filters followed by a PbSe detector for the 1.5-6 μ band, and a Ge:Hg detector for the 4-15 μ band for $\lambda/\Delta\lambda = 100$. o A combination of radiation cooling and an open-cycle N ₂ /He cooling system for cooling the PbSe detector to 130°K and the Ge:Hg to approximately 27°K. Total operation time at 27°K is 30 minutes.		<u>REMARKS:</u> * Based on six bits per spectral element, at 200 elements per second plus 130 bits for auxiliary data concerning location of source of radiation, wavelength, detector temperature monitor, etc.				
<u>EXPERIMENT REQUIREMENTS:</u> o System to be directed at areas of interest, normally within field of imaging equipment. o System should not view Sun directly.						

Table 4-5: EXPERIMENT SUMMARY SHEET

EXPERIMENT NUMBER No. 3a	ATMOSPHERIC TEMPERATURE SOUNDING EXPERIMENT (HIGH RESOLUTION INFRARED SPECTROMETER)	PHYSICAL AND FUNCTIONAL CHARACTERISTICS			
			1973	1975	1977
			1979		
EXPERIMENT OBJECTIVES:		Weight (pounds) Power (watts) Volume (cubic inches) Data (bits/second) Look Angles (deg) Cone Clock Field of view (deg)			
1973:		30 14 2400 1000 80-180 Along nadir 1.1 x 4			
1. To determine the vertical temperature variation from measure of the thermal emission from 4.3 μ CO ₂ band.					
2. To determine surface (or cloud top) temperature from measure of thermal emission in the 4.5 to 5.0 μ band.					
3. To determine planet reflectance using 1.5 to 2.0 μ and 3.0 to 4.0 μ bands in conjunction with reflected sunlight.					
4. To estimate the H ₂ O present by measuring near 2.5 μ .					
5. To determine whether high clouds are present by detecting radiation near 2.7 μ .					
INSTRUMENT DESIGN CHARACTERISTICS:					
1973:					
o Modified Ebert design spectrometer with an f/3, 10-inch diameter collector optics, a 250 line/mm grating, radiative cooled PbSe and PbS detectors (to 213°K). To cover bands 3-5 μ and 1-3.5 μ , respectively, with spectral resolutions $\lambda/\Delta\lambda = 100$ at 4.3 μ and $\lambda/\Delta\lambda = 250$ at 2.5 μ .					
o First order spectra in 3-5 μ band and second order spectra for 1.0-3.5 μ band are separated by dichroic mirror.					
EXPERIMENT REQUIREMENTS:					
o Instrument assembly to be mounted on platform out of direct sunlight on thermal isolation mounts.					
o A clear aperture of 8 in ² is required for radiative cooling - at no time may sunlight strike this radiating area.					
o At no time should the instrument's entrance aperture be pointed directly at the Sun.					
o Direction of travel along 1.1° field direction.					
o Altitude at time of measurement less than 20,000 km.					
o Sun angle at point of measurement: 0-100°.					
REMARKS:					

Table 4-6: EXPERIMENT SUMMARY SHEET

EXPERIMENT NUMBER No. 3b	HIGH RESOLUTION INFRARED SPECTROMETER	PHYSICAL AND FUNCTIONAL CHARACTERISTICS			
		1973	1975	1977	1979
<u>EXPERIMENT OBJECTIVES:</u> <u>1975:</u> Combined objectives of the atmospheric sounding experiment and the broadband infrared spectrometer (see Experiments No. 2 and 3a). <u>1977:</u> Same as 1975. <u>1979:</u> Different orbital parameters			41 20 3000 1500 80-180 Along nadir 1.1x4	41 20 3000 1500 80-180 Along nadir 1.1x4	41 95 3000 1500 80-180 Along nadir 1.1x4
<u>INSTRUMENT DESIGN CHARACTERISTICS:</u> <u>1975:</u> Modified Ebert design spectrometer with an $f/3$, 10-inch diameter collector optics, and 500 line/mm grating. Operation in the spectral bands $1-3.5 \mu$, $3-5 \mu$, and $8-12 \mu$ using radiative cooling (to -60°C) for PbS and PbSe, respectively, for the first bands and Ge:Hg cooled by an open cycle (N_2/He) cooler (approximately 10 lb for 30 minutes operation) to 35°K for the $8-13 \mu$ band. The spectral resolution would be $\lambda/\Delta\lambda \approx 250$ for the $1-3.5 \mu$ band and $\lambda/\Delta\lambda \approx 100$ for the other bands. <u>1977:</u> Same as 1975. <u>1979:</u> Same as in 1977, except that the mid-infrared spectral range operates continuously in the $8-10 \mu$ band using a closed-cycle cooler (approximately 10 lb and 75 watts) to cool a Ge:Au detector to 77°K .					
<u>EXPERIMENT REQUIREMENTS:</u> Same as Experiments No. 2 and 3a.					
					REMARKS :

Table 4-7: EXPERIMENT SUMMARY SHEET

EXPERIMENT NUMBER No. 4	INFRARED RADIOMETER (STRIP MAPPER)	PHYSICAL AND FUNCTIONAL CHARACTERISTICS				
		1973	1975	1977	1979	
EXPERIMENT OBJECTIVES:	1973:					
	o To determine the temperature of Mars					
	- Temperature variation in the area of the terminator to yield information concerning composition of the surface.					
	- Local temperature variations on the surface temp. resulting from diurnal cycle and anomalies.					
	o To aid in identifying cloud features by temperature measurement.					
1975:						
o To obtain higher resolution strip maps using thermal emission from the planet's surface to locate regions of volcanic activity or other anomalous thermal activity.						
o To obtain strip maps of H ₂ O and CO ₂ temperature and distribution.						
o Possibly to measure temperatures of Deimos and Phobos.						
1977:	Same objectives as 1975, except for different geographic regions.					
1979:						
INSTRUMENT DESIGN CHARACTERISTICS:						
1973:						
Rotating mirror of 5 in diameter aperture scans $\pm 60^\circ$ strip during the period $\pm 60^\circ$ of the perigais. Collected flux is divided between two detectors covering 8-12 μ and 18-25 μ spectral bands.						
1975-1979:						
Strip mapper uses a linear array of 10 detectors aligned parallel to the scanner's velocity vector. A 7-inch diameter aperture system with object field scanning.						
EXPERIMENT REQUIREMENTS:						
1973-1979:						
Mounting on a 3-axis planetary scan platform coincident with the field of view of the photo imaging device and such that the direction of the mirror's scan is perpendicular to the orbit plane. System should not view the Sun directly.						
REMARKS:		Dr. Richard Shorthill of BSRL suggests that the first mission should be of a survey type giving data on the entire planet at the expense of resolution. Subsequent missions should have greatly improved resolution and be concerned primarily with areas of interest located during the survey mission. Correlation between the thermal mapping and photomapping is important since in many cases a correlation is possible.				

Table 4-8: EXPERIMENT SUMMARY SHEET

EXPERIMENT NUMBER No. 5, No. 16	ULTRAVIOLET SPECTROMETER AND SOLAR OCCULTATION	PHYSICAL AND FUNCTIONAL CHARACTERISTICS				
		1973	1975	1977	1979	
<u>EXPERIMENT OBJECTIVES:</u>						
<u>1973:</u>						
To detect atoms, ions, and molecules in the upper atmosphere, measure scale heights and measure lower atmosphere scattering densities and surface reflectance. Both day and night observations will be made and seasonal and latitudinal effects sought.						
<u>1975:</u>						
Continue examination of seasonal and latitude effects, increase spectral resolution, increase spatial resolution of sampling, increase sensitivity for nightglow and auroral measurements plus Mars corona detection.						
<u>1977:</u>						
Extend spectral range to 300 Å for UV (solar occultation)						
<u>INSTRUMENT DESIGN CHARACTERISTICS:</u>						
<u>1973:</u>						
Fastio-Ebert spectrometer similar to Mariner '69, resolution about 20 Å, spectral range 1100-4000 Å.						
<u>1975:</u>						
Increase telescope focal length on '73 instrument, decrease entrance slit size, increase detector channels from 2 to 4 and reduce scan speed and range.						
<u>1977:</u>						
Far UV spectrometer with multidetector channels, use Sun as light source.						
<u>EXPERIMENT REQUIREMENTS:</u>						
<u>1973:</u>						
Slit parallel to limb ± 3 degrees. Temperature control across secondary mirror support structure to keep differential below 20°K. Operating temperatures 270-310°K. Measurements from 2500-km altitude. All latitudes. Summer and winter seasons.						
<u>1975:</u>						
Same as 1973, except that measurements are taken at 500-km altitude.						
<u>1977:</u>						
Slit parallel to limb ± 8 degrees. Telescope must image Sun on entrance slit with accuracy of 0.3 degrees and hold this each data-taking period with a stability of 0.1 degree. Altitude for measurements is 1000 km.						
		Weight (pounds) Power (watts) Volume (cubic feet) Data (bits/second) (total bits/orbit) (total bits/mission) Operating time Look Angles Cone (deg) Field of view (deg)				
		30 12 0.8 4000 1.5x10 ⁵ 3x10 ⁷ 10 min/orbit 60 orbits 90-105 1x1	40 20 1.5 8000 7x10 ⁵ 1.5x10 ⁸ 10 min/orbit 30 min/orbit 90-105 1x1	30 25 2 4000 1.5x10 ⁵ 10 ⁷ 30 min/orbit 0 1x1		
		<u>REMARKS:</u> 1973-1975: Spectrometer pointed towards illuminated limb of planet approximately normal to Sun axis, slit parallel to limb. 1977: Spectrometer pointed towards Sun at passage into Mars shadow (occultation of Sun by Mars). Trajectory should pass through center of Mars shadow cone, if possible.				

Table 4-9: EXPERIMENT SUMMARY SHEET

EXPERIMENT NUMBER No. 6, No. 9	PLASMA PROBE AND TRAPPED RADIATION DETECTOR	PHYSICAL AND FUNCTIONAL CHARACTERISTICS				
		1973	1975	1977	1979	
<u>EXPERIMENT OBJECTIVES:</u> 1973: Plasma Probe & Trapped Radiation Detector - Measure solar wind properties (energy, flux) in interplanetary space. Monitor solar events. Determine existence of magnetosphere and trapped radiation belts at Mars. Ordered spatial variations in plasma or trapped radiation fluxes would indicate the presence of a magnetosphere or trapped radiation belts and enhance the effectiveness of the subsatellite experiment. 1975: None. See Remarks. 1977: None 1979: None		<u>SPACECRAFT INTERFACES</u> Weight (pounds) Power (watts) Volume (cubic inches) Data (bits per second) Operating Time Plasma Probe 10 3 228 23.0 Cont. Trap. Rad. 3 80 15.5 Cont.				
<u>INSTRUMENT DESIGN CHARACTERISTICS:</u> 1973: Plasma Probe - Curved plate analyzer (Mariner type) Trapped Radiation Detector - (Mariner IV type) 1975: Plasma Probe, Trapped Radiation Detector, and Magnetometer are contained within a subsatellite, which is ejected from the S/C after Mars orbit is obtained. No measurements are possible during cruise. Plasma and trapped radiation instruments should be more sophisticated in 1975 than in 1973. Subsatellite carries power supply and telecommunications equipment.		<u>REMARKS:</u> Magnetometer cannot be carried on S/C because of lack of magnetic cleanliness; thus, it is proposed as a subsatellite experiment. In the absence of a magnetometer, complex plasma probe and trapped radiation detector are not warranted in 1973. However, these instruments should be flown in conjunction with the magnetometer in the subsatellite. See Experiment 13.				
<u>EXPERIMENT REQUIREMENTS:</u>						

Table 4-10: EXPERIMENT SUMMARY SHEET

EXPERIMENT NUMBER No. 7, No. 8, No. 10		COSMIC RAY TELESCOPE, COSMIC DUST DETECTOR, AND ION CHAMBER	PHYSICAL AND FUNCTIONAL CHARACTERISTICS					
			1973		1975	1977	1979	
<u>EXPERIMENT OBJECTIVES:</u>			Cos. Ray	Cos. Dust	Ion Chamber	Cos. Ray	Ion Chamber	Cosmic Dust Dete.
1973: Ion Chamber - Monitor intermediate energy protons associated with solar particle events (10-50 Mev range). Cosmic Ray Telescope - Measure spectrum of energetic galactic cosmic rays. The effects of solar cycle variation and the gradient of galactic cosmic rays from 1 to 1.5 A.U. are of particular interest. Cosmic Dust Detector - Measure cosmic dust flux as a function of direction, distance from Sun, momentum and distribution about planet.			8	5	3	8	3	5
1975: Continue ion chamber and cosmic ray telescope experiments.			3	1	1	3	1	1
1977: Continue cosmic dust detector experiment.			624	118	82			
			46	1.6	15.5	46	15.5	1.5
			Continuous					
<u>INSTRUMENT DESIGN CHARACTERISTICS:</u>								
1973: Ion Chamber - Neher & Anderson (Mariner IV) Cosmic Ray Telescope - U of Chicago instrument Cosmic Dust Detector - Combination capacitor penetration and microphonic instrument mounted perpendicular to solar panels and to ecliptic plane.								
1975: Ion Chamber - Cosmic Ray Telescope - more channels than 1973 instrument.								
1977: Cosmic Dust Detector								
<u>EXPERIMENT REQUIREMENTS:</u>								
<u>REMARKS:</u>								

Table 4-11: EXPERIMENT SUMMARY SHEET

EXPERIMENT NUMBER		ATMOSPHERIC POLARIMETER	PHYSICAL AND FUNCTIONAL CHARACTERISTICS			
No. 11			1973	1975	1977	1979
<u>EXPERIMENT OBJECTIVES:</u> <u>1973:</u> To obtain size distribution, number density, and composition of particulate component of haze, dust, clouds and droplets in Martian atmosphere. A measurement will be made of wavelength variation and polarization of scattered sunlight. <u>1977:</u> Extend surface coverage and seasonal changes, increase spectral regions covered from 3 to 6 wavelengths.			9 3 0.3 3 x 10 ² 40 min/orbit Nadir 140 deg in orbit plane 0.5 deg cross plane		9 3 0.3 6 x 10 ² 40 min/orbit Nadir 140 deg in orbit plane 0.5 deg cross plane	
<u>INSTRUMENT DESIGN CHARACTERISTICS:</u> Spectropolarimeter: An image dissector tube will be used with spectral filters and polarizers at three wavelength regions. A rotating mirror will be used to scan a swath 1 1/2 degrees wide by 140 degrees long in the direction of motion. Mirror drive may be pressurized for lubricant preservation. A ground resolution of 10 to 20 km will be obtained, with vertical resolution at the horizon perhaps 2 to 3 times greater.			<u>REMARKS:</u>			
<u>EXPERIMENT REQUIREMENTS:</u> Exhaust gases from spacecraft must not get in field of view. Data storage will be needed because of intermittent data acquisition cycle. Ground track of spacecraft should pass over regions photographed so polarimeter data should be correlated with images.						

Table 4-12: EXPERIMENT SUMMARY SHEET

EXPERIMENT NUMBER No. 12	ATMOSPHERIC MASS SPECTROMETER	PHYSICAL AND FUNCTIONAL CHARACTERISTICS	1973	1975	1977	1979
EXPERIMENT OBJECTIVES: <u>1973:</u> To measure some of the Martian upper atmosphere constituents.		Weight (pounds) Power (watts) Volume (cubic inches) Data (bits per second) per orbit Operating Time Look Angles	8 10 160 60 3000 1 sec./sweep Along orbital path			
	INSTRUMENT DESIGN CHARACTERISTICS: A GE mass spectrometer of monopole design will cover a range from 0 to 50 AMU at a rate of one sweep per second with a sensitivity of 1 to 10 parts per million and a resolution of 5% of peak. Logarithmic gain control for the determination of existing elements in sequence would be more appropriate.					
	EXPERIMENT REQUIREMENTS: The instrument operates by sampling the atmosphere continuously; thus, the sampling port must point in the direction of the orbital path and exhaust in the trail to prevent sampling of the continuation of the internal instrument chamber.					
		REMARKS:				

Table 4-13: EXPERIMENT SUMMARY SHEET

EXPERIMENT NUMBER No. 13	SATELLITE EXPERIMENT	PHYSICAL AND FUNCTION CHARACTERISTICS				1973	1975	1977	1979
<u>EXPERIMENT OBJECTIVES:</u> 1975: 1. Make precision measurements of the Mars Magnetospheres and/or bow shock wave and turbulent wake using a magnetometer and suitable charged particle detectors (solar plasma and soft electronics). 2. Measure the solar plasma and soft electron spectra and flux in areas outside the shock wave or wake regions. 3. Conduct electromagnetic probing of the Mars atmosphere using a bistatic link between the subsatellite and the orbiter. 1979: Same as 1975		Weight (pounds) Power (watts) Volume (cubic feet) Data (bits per second)					160 5 30 500		160 5 30 500
<u>INSTRUMENT DESIGN CHARACTERISTICS:</u> Instrument will be a spin-stabilized solar-powered probe carrying the following instruments: Magnetometer Plasma probe Medium energy particle detector Multi frequency radio beacon transmitter In addition, a simple data sequencer, data encoder and telemetry transmitter will provide for transmission of data from the subsatellite to the spacecraft. Spin-up propulsion and launching mechanism to impart approximately 10 rpm and 15 fps will also be required. The spacecraft terminal equipment is listed under Experiment No. 14 (Bistatic Experiment).		<u>REMARKS:</u>							
<u>EXPERIMENT REQUIREMENTS:</u> Subsatellite should be launched in plane of orbit at a relative speed of 15 fps, so that bistatic experiment will begin about 30 days after subsatellite launch.									

Table 4-14: EXPERIMENT SUMMARY SHEET

EXPERIMENT NUMBER No. 14	BISTATIC EXPERIMENT	PHYSICAL AND FUNCTIONAL CHARACTERISTICS				1973	1975	1977	1979
<p><u>EXPERIMENT OBJECTIVES:</u></p> <p>1975: Conduct electromagnetic probing of the Mars atmosphere in conjunction with the subsatellite experiment (No. 13).</p> <p>1977: Same as 1975 except that experiment will be performed in conjunction with second orbiter; also, relative polarization of the received waves will be measured.</p> <p>1979: Same as 1975.</p>		<p><u>INSTRUMENT DESIGN CHARACTERISTICS:</u></p> <p>1975: This is the spacecraft (receiver terminal of the electromagnetic) probing experiment. Receiver is a multi-frequency receiver operating in the neighborhood of 50 MHz and 450 MHz. Two whip antennas will be used.</p> <p>1977: Same as 1975 except that capability to measure polarization of received signals will be added. Two whip antennas 90 degrees apart will be used for each frequency.</p> <p>1979: Same as 1975</p>				<p><u>REMARKS:</u></p> <p>The 1975 mission will require precise timing of both orbiters to ensure that the line of sight between the two spacecraft will pass through the Mars atmosphere early in the mission.</p>			
<p><u>EXPERIMENT REQUIREMENTS:</u></p> <p>1975: Antennas will be located parallel to Mars-Sun line.</p> <p>1977: The two antennas for each frequency will be located perpendicular to each other. Relative orientation of antennas on both spacecraft must be mirror images of each other and antennas must be perpendicular to line of sight between two spacecraft at time of occultation.</p> <p>1979: Antennas mounted as in 1975.</p>									

Table 4-15: EXPERIMENT SUMMARY SHEET

EXPERIMENT NUMBER No. 15	BISTATIC RADAR	PHYSICAL AND FUNCTIONAL CHARACTERISTICS				
		1973	1975	1977	1979	
EXPERIMENT OBJECTIVES: 1977: Measure the radar returns from the Mars surface to determine surface and subsurface roughness characteristics, major continental features, and possibly the locations of subsurface water deposits. The greater penetration of radio frequency waves will give additional information about the nature of the underlying structure.		Weight (pounds) 20 Power (watts) 20 Volume (cubic feet) 0.5 Data (bits per second) 4000 Antenna Pointing Angle Along nadir				
INSTRUMENT DESIGN CHARACTERISTICS: Spacecraft instrument will be a 200 MHz receiver for either CW (continuous wave) or long pulse (~100µsec) reception. Transmitter will be Earth-based (1 Mw) operating into a high gain antenna. Receiver measures Doppler characteristic of signal returned from Mars surface. Polarization in two orthogonal planes will be measured. Moderate antenna gain will be required to increase signal to noise ratio.		REMARKS: Data should be correlated with visual and IR scanner information.				
EXPERIMENT REQUIREMENTS: Antenna points toward Mars surface. One plane of polarization should be parallel to incident wave from Earth.						

Table 4-16: EXPERIMENT SUMMARY SHEET

EXPERIMENT NUMBER No. 17	GAMMA RAY SPECTROMETER	PHYSICAL AND FUNCTIONAL CHARACTERISTICS			
		1973	1975	1977	1979
EXPERIMENT OBJECTIVES: 1975: Measure spectra of natural or induced gamma ray activity of Mars surface to obtain evidence on composition, differentiation, and thermal history. 1977: Increase detector sensitivity, energy resolution, and improve spatial resolution and surface coverage.	INSTRUMENT DESIGN CHARACTERISTICS: 1975: A NaI (Tl) crystal scintillator with plastic scintillator charged particle shield and 256 channel analyzer. The detector is mounted on a boom at least 10 feet and preferably 20 feet from spacecraft; the analyzer is in the spacecraft. Energy 0.1-10 Mev spectra. 1977: Use a larger detector with an active collimator shield to improve spatial resolution and sensitivity and a 512-channel analyzer to increase energy resolution.	Weight (pounds)	15	20	
		Power (watts)	8	12	
		Volume (cubic feet)	0.5	0.8	
		Data (bits per second)	50	200	
		Operating Time	Continuous	Continuous	
EXPERIMENT REQUIREMENTS: Radioactive cleanliness on spacecraft, especially Th, K, U. Isolation from secondary radiation, produced by cosmic ray bombardment of spacecraft, by utilizing a 10 - 20-foot boom. Boom must face planet with unobstructed view of entire surface. Extension of boom before planetary arrival is needed to obtain background measurements. Planet oriented platform required. Low Altitude - down to 500 km - desired	EXPERIMENT REQUIREMENTS: Radioactive cleanliness on spacecraft, especially Th, K, U. Isolation from secondary radiation, produced by cosmic ray bombardment of spacecraft, by utilizing a 10 - 20-foot boom. Boom must face planet with unobstructed view of entire surface. Extension of boom before planetary arrival is needed to obtain background measurements. Planet oriented platform required. Low Altitude - down to 500 km - desired	Look Angles	See expmt. reqmts.	See expmt. reqmts.	
		Cone	150	180	
		Clock			
		Field of view (deg)			
		REMARKS:			

Table 4-17: EXPERIMENT SUMMARY SHEET

EXPERIMENT NUMBER No. 18	NEUTRON ALBEDO DETECTOR	PHYSICAL AND FUNCTIONAL CHARACTERISTICS	1973	1975	1977	1979
<p>EXPERIMENT OBJECTIVES:</p> <p>1979: Determine the neutron flux produced by cosmic ray interaction with atmospheric and crust constituents. These measurements will be correlated with trapped radiation measurements and magnetometer readings to determine general properties of the Martian atmosphere and crust.</p> <p>INSTRUMENT DESIGN CHARACTERISTICS:</p> <p>Two types of BF₃ counters (with different isotopic enrichments) surrounded by a hydrogenous moderator. The background due to highly ionizing events can be determined by comparing the two instrument readings.</p> <p>An anticipated counting rate at 1000 km may be from 10^{-1} to 10^2/sec.</p> <p>EXPERIMENT REQUIREMENTS:</p> <p>Separate detector from spacecraft as far as possible to reduce background due to cosmic-ray-induced neutrons in the spacecraft. Trapped radiation detector must be available to determine if any Martian radiation belts are affecting the results.</p> <p>Counting rates may range upward several orders of magnitude during solar particle events. Peak counting rates will be at periaapsis which should be as low as possible. A highly eccentric orbit would be necessary to assess the influence of the spacecraft-induced neutrons.</p> <p>Altitude at periaapsis as low as possible</p>		<p>Weight (pounds) 30</p> <p>Power (watts) 1</p> <p>Volume (cubic feet) 0.8</p> <p>Data (bits per second) 1</p> <p>Operating Time 3 hr/orbit</p> <p>Stability (deg) ± 20</p> <p>Look Angle Nadir</p>				
<p>REMARKS:</p> <p>Location of the experiment on the subsatellite might help to reduce the space vehicle component of cosmic-ray-induced neutrons.</p> <p>Reference: JGR 67, 499 1962 OSO-1 experiment (DASA survey, Part 2)</p>						

Table 4-18: EXPERIMENT SUMMARY SHEET





EXPERIMENT NUMBER No. 19	METEOR FLUX DETECTOR	PHYSICAL AND FUNCTIONAL CHARACTERISTICS				1973	1975	1977	1979
<p><u>EXPERIMENT OBJECTIVES:</u></p> <p>1977: Measure flux of meteors entering Mars atmosphere with sufficient energy to become incandescent. The intensity and duration of the light pulse measured at the spacecraft will give information as to the particle mass. Velocity can be inferred from the pulse characteristics.</p>		<p>Weight (pounds)</p> <p>Power (watts)</p> <p>Volume (cubic inches)</p> <p>Data (bits per second)</p> <p>Look Angle</p> <p>Field of view (deg)</p>						<p>6</p> <p>5</p> <p>290 </p> <p>10</p> <p>Along nadir 30 </p>	
<p><u>INSTRUMENT DESIGN CHARACTERISTICS:</u></p> <p>Instrument consists of a wide angle photometer operating in the near-infrared, pointed along the spacecraft nadir towards the dark side of Mars. Instrument sensitivity will be reduced when spacecraft is on sunlit side or near terminator so that operation will be effective only when spacecraft is approximately 10 deg beyond terminator.</p>						<p><u>REMARKS:</u></p> <p> Based on 60 bits/measurement, one measurement every 6 seconds.</p> <p> Field of view is tentatively estimated at 30 degrees. This parameter should be optimized for each specific orbit to maximize the time integral of area viewed on the dark side of Mars.</p>			
<p><u>EXPERIMENT REQUIREMENTS:</u></p> <p>Instrument should be mounted on back of scan platform. No scattered light is permitted within a 40 degree field of view when the instrument is pointed along the spacecraft nadir at cone angles from 0 to 80 degrees.</p>									

Table 4-19: EXPERIMENT SUMMARY SHEET

EXPERIMENT NUMBER No. 20a	SOLAR AND CELESTIAL X-RAY MAPPING	PHYSICAL AND FUNCTIONAL CHARACTERISTICS			
		1973	1975	1977	1979
<u>EXPERIMENT OBJECTIVES:</u> 1977: 1) To provide long-term mapping of the Sun at X-ray wavelengths, particularly to determine variations in solar x-radiation with longitude. 2) To measure time variations of selected x-ray sources.	<u>INSTRUMENT DESIGN CHARACTERISTICS:</u> Grazing incidence X-ray telescope with angular resolution of ≈ 10 arc sec and collecting area of 10 cm^2 mounted on scan platform with 3-axis control. Wavelength range of 5 to 500 Å (0.24 to 2.4 kev). Knowledge of direction of pointing needed to ± 5 arc sec. Cylindrical package of 10 cm diameter and 100 cm length on '77 mission. Resolution will increase with increasing payload to ≈ 5 arc sec with larger instrument on later missions.	Weight (pounds) Power (watts) Volume (cubic feet) Data (bits per second) Operation Time Stability		15 10 1 1.5×10^3 50% during planetary	
				10 arc sec/min	
		<u>REMARKS:</u> This type of experiment is worthwhile, but does not fit in with the objectives of planetary exploration. X-ray astronomy can generally be done as well from an Earth orbital satellite. Solar mapping from different solar longitudes may be somewhat more appropriate, especially if other vehicles to go into independent solar orbit are scarce. Hence, this experiment could be added to a Voyager payload as a low priority experiment.			
		<u>EXPERIMENT REQUIREMENTS:</u> Pointing accuracy of 5 arc sec will require special platform stabilization Unobstructed view for scan platform over field of at least one octant. Location as far as possible from large masses in the S/C to minimize cosmic ray secondaries.			

Table 4-20: EXPERIMENT SUMMARY SHEET

EXPERIMENT NUMBER No. 20b	CELESTIAL X-RAY BACKGROUND	PHYSICAL AND FUNCTIONAL CHARACTERISTICS				1973	1975	1977	1979
EXPERIMENT OBJECTIVES: 1977: (Alternate) To measure the diffuse X-ray background with regard to intensity and isotropy and to resolve discrete sources in the background.		Weight (pounds) Power (watts) Volume (cubic feet) Data (bits per second) Operating Time Stability Look Angle Field of view				20 2 to 4 1.0 2×10^2 Cont. during cruise 50% during planetary orbit 30 arc sec/min- also a scan mode of 5 arc min/sec Anti-Sun 20 x 20 deg			
INSTRUMENT DESIGN CHARACTERISTICS: Large area ($\sim 10^3 \text{ cm}^2$) proportional counter with grid of wires to provide resolution with modulation collimators. (Angular resolution ~ 30 arc sec). 8 to 16 channel pulse height analysis could be done on X-rays in wavelength range 0.4 Å to 10 Å (1.2 to 30 kev). Package size on '77 mission = 1 ft x 1 ft x 1 ft including electronics.		REMARKS: This type of experiment, though worthwhile, does not fit in with the primary, planetary exploration objectives of the Voyage and has no particular reason for being included on a planetary mission. X-ray astronomy can be done as well from an Earth orbital satellite. However, this experiment could be included as a low priority experiment if weight and power are available.							
EXPERIMENT REQUIREMENTS: Location as far as possible from large masses in the S/C to minimize cosmic ray secondaries. Unobstructed view for scan platform (if needed to provide coverage of total sky) of approximately one octant.									

Table 4-21: EXPERIMENT SUMMARY SHEET

EXPERIMENT NUMBER No. 21		IONOSPHERE SOUNDER	PHYSICAL AND FUNCTIONAL CHARACTERISTICS				
<u>EXPERIMENT OBJECTIVES:</u> 1975: Determine ionosphere electron density profile as a function of latitude and sun angle. As an objective, measurements should be made approximately every 2 degrees. 1977: Same as 1975 - for different latitudes and Sun angles.					1975	1977	1979
					25	(Same as 1975)	
					10		
<u>INSTRUMENT DESIGN CHARACTERISTICS:</u> Sounder is a fixed frequency pulsed sounder equipped with two orthogonal dipole antennas, one 40 feet and the other 120 feet from tip to tip. Sounder makes a complete sounding in 40 seconds up to a range of 7000 km and in 240 seconds up to a range of 17,200 km.					100		
					<u>REMARKS:</u>		
<u>EXPERIMENT REQUIREMENTS:</u> Antennas are best located at an angle of 45 degrees from plane of initial orbit.							

Table 4-22: EXPERIMENT SUMMARY SHEET

EXPERIMENT NUMBER No. 22		EARTH OCCULTATION	PHYSICAL AND FUNCTIONAL CHARACTERISTICS				
EXPERIMENT OBJECTIVES:			1973	1975	1977	1979	
1973: By observing the decay in the S-band telemetry signal of the spacecraft power into and out of Earth occultation:							
1. Improve knowledge of the density and pressure profiles of the Martian atmosphere and determine how these profiles vary with Sun angle.							
2. Improve knowledge of the electron density profiles of the Martian ionosphere and determine how these vary with latitude and Sun angle.							
3. Improve knowledge of the diameter and figure of Mars.							
1975: { Same as 1973 to obtain information at different Sun angles and latitudes,							
1977: { also to determine statistical variation of these parameters.							
1979: }							
INSTRUMENT DESIGN CHARACTERISTICS:							
Experiment will utilize the existing S-band telemetry on the spacecraft.							

Table 4-23: EXPERIMENT SUMMARY SHEET

EXPERIMENT NUMBER No. 23		CELESTIAL MECHANICS EXPERIMENT	PHYSICAL AND FUNCTIONAL CHARACTERISTICS				
EXPERIMENT OBJECTIVES:			1973	1975	1977	1979	
1973: To establish and improve knowledge of the following parameters: 1. The Mars gravitational harmonic coefficients 2. Ephemeris of the Mars orbit 3. The mass of Mars 1975: Same as 1973 1977: 1979:			0	0	0	0	
			0	0	0	0	
			0	0	0	0	
			0	0	0	0	
INSTRUMENT DESIGN CHARACTERISTICS: This experiment will use the DSN tracking facility. No additional on-board equipment will be required.			REMARKS: This experiment requires no additional on-board equipment. It is included since it will have an impact on orbit design parameters.				
EXPERIMENT REQUIREMENTS: A desirable orbit for this experiment would: 1. Permit Earth occultation as little as possible during the early part of the mission. 2. Have an orbit inclination to the Mars equator that is well away from 0 or 90 degrees. 3. Be highly eccentric. 4. Maintain telemetry communication with Earth for as large a percentage of time as possible.							

4.3.7 Science Payload Impact

The science payload affects the spacecraft and mission. This results from the following key factors:

- 1) The payload weight and size
- 2) The payload power needs
- 3) The data acquired by the science payload.

These factors are summarized in Table 4-24 for the payloads proposed for the 1973-1979 mission.

The following science payload requirements also affect the spacecraft and mission:

- 1) viewing angles,
- 2) orbit characteristics,
- 3) sequencing,
- 4) stabilization,
- 5) modularization,
- 6) thermal control, and
- 7) electromagnetic interference (EMI) considerations.

Detailed science payload power requirements are given in Table 4-25. The expected power consumption and time of operation for each instrument for a typical orbit are given. The power profiles for each mission are shown in Figure 4-13. The peak power load grows from 163 watts in 1973 to 223 watts in 1975 and to about 370 watts in 1977 and 1979.

The science payload data acquisition rates are given in Table 4-26 for the candidate payloads for the four missions. During the typical orbit shown, all scanning instruments operate at the same time to permit correlation of their data. The photoimaging experiment generates most of the science data. The science payload generated data profile evolution is shown in Figure 4-14.

Table 4-24: PROPOSED 1973-1975-1977-1979 SCIENCE PAYLOAD EXPERIMENTS AND CHARACTERISTICS

MISSION EXPERIMENT	1973			1975			1977			1979		
	Weight (lb)	Power (watts)	Data* (bits/sec)	Weight (lb)	Power (watts)	Data* (bits/sec)	Weight (lb)	Power (watts)	Data* (bits/sec)	Weight (lb)	Power (watts)	Data* (bits/sec)
Imaging	150	60	5×10^5	165	100	5×10^5	660	200	2.5×10^7	660	200	2.5×10^7
Broadband IR Spectrometer	25	5	1330/665									
High Resolution IR Spectrometer	30	14	1000	41	20	1500	41	20	1500	41	95	1500
IR Radiometer	20	6	2300	25	10	46,000	25	10	46,000	25	10	46,000
UV Spectrometer	30	12	4000	40	20	8,000						
Plasma Probe	10	3	23	(In Subsatellite)						(In Subsatellite)		
Cosmic Ray Telescope	8	3	46	8	3	46						
Cosmic Dust Detector	5	1	1.6				5	1	1.6			
Trapped Radiation Detector	3	1	15.5	(In Subsatellite)						(In Subsatellite)		
Ion Chamber	3	1	15.5	3	1	15.5						
Atmospheric Polarimeter	9	3	300				9	3	600			
Atmospheric Mass Spectrometer	8	10	60									
Subsatellite ****				160	5	500				160	5	500
Bistatic Experiment**				40	2	10	45	2	10	40	2	10
Bistatic Radar***							20	(20)	4000			
Solar Occultation							30	25	4000			
Gamma Ray Spectrometer				15	8	50	20	12	200			
Neutron Albedo												
Meteor Flux							6	5	10	30	1	1
Celestial X-Ray							15	10	1500			
Ionosphere Sounder				25	10	100	25	10	100			
Earth Occultation***	0	0	0	0	0	0	0	0	0	0	0	0
Celestial Mechanics	0	0	0	0	0	0	0	0	0	0	0	0
Subtotal Science Instruments	301	119		522	179		901	298		956	313	
DAE And Misc.	70	44		70	44		100	54		100	54	
Total Science Payload	371	163		592	223		1001	352		1056	367	

* Average During Data Acquisition Period

** Spacecraft — Subsatellite Or Spacecraft — Spacecraft Experiment

*** Earth — Spacecraft Experiment

**** Includes Plasma Probe, Trapped Radiation Detector, Bistatic Experiment and Magnetometer


 Bistatic Radar Off During Peak Power

Table 4-25: DETAILED SCIENCE PAYLOAD POWER REQUIREMENTS

Mission Experiments	1973		1975		1977		1979	
	Power (watts)	Time of Oper. (min)	Power (watts)	Time of Oper. (min)	Power (watts)	Time of Oper. (min)	Power (watts)	Time of Oper. (min)
Imaging	60	-54 to -5	100	-54 to -5	200	-54 to -5	200	-54 to -5
Broadband IR Spectrometer	5	-59 to +10	20	-59 to +10	20	-59 to +10	95	-59 to +10
Hi Res. IR Spectrometer	14	-59 to 0	10	-59 to +10	10	-59 to +10	10	-59 to +10
IR Radiometer	6	-59 to +10	20	-59 to 0	2	-59 to +10		
UV Spectrometer	12	-59 to 0	20	-59 to 0				
Plasma Probe	3	Continuous	3	Continuous	1	Continuous		
Cosmic Ray Telescope	3	Continuous	1	Continuous	3	-65 to +10		
Cosmic Dust Detector	1	Continuous	5	Continuous	2	Continuous	5	Continuous
Trap. Radiation Detector	1	Continuous	2	Continuous	(20)	-80 to -20	2	Continuous
Ion Chamber	1	Continuous	8	Continuous	25	-10 to +50		
Atmos. Polarimeter	3	-65 to +10			12	Continuous	1	Continuous
Atmos. Mass Spectrometer	10	-45 to +10			12	Continuous		
Subsatellite					5	Continuous		
Bistatic Experiments					10	Continuous		
Bistatic Radar					29	Continuous		
Solar Occultation					10	Continuous		
Gamma Ray Spectrometer					29	Continuous		
Neutron Albedo					25	Continuous		
Meteor Flux								
Celestial X-ray								
Ionosphere Sounder								
DAE & Power Switch Elec.	29	Continuous	10	Continuous				
Scan Platform No. 1	10	-59 to +10	5	-59 to 0				
Scan Platform No. 2	5	-59 to 0						
Standby Power, Lge. Camera								
Total Power Required	163		223		352		367	

▷ Time in minutes before and after evening terminator; includes 30-minute warmup time.

▷ Alternate mode: UV spectrometer operates from 30 min before to 60 min after evening terminator

▷ Bistatic radar off during peak power.

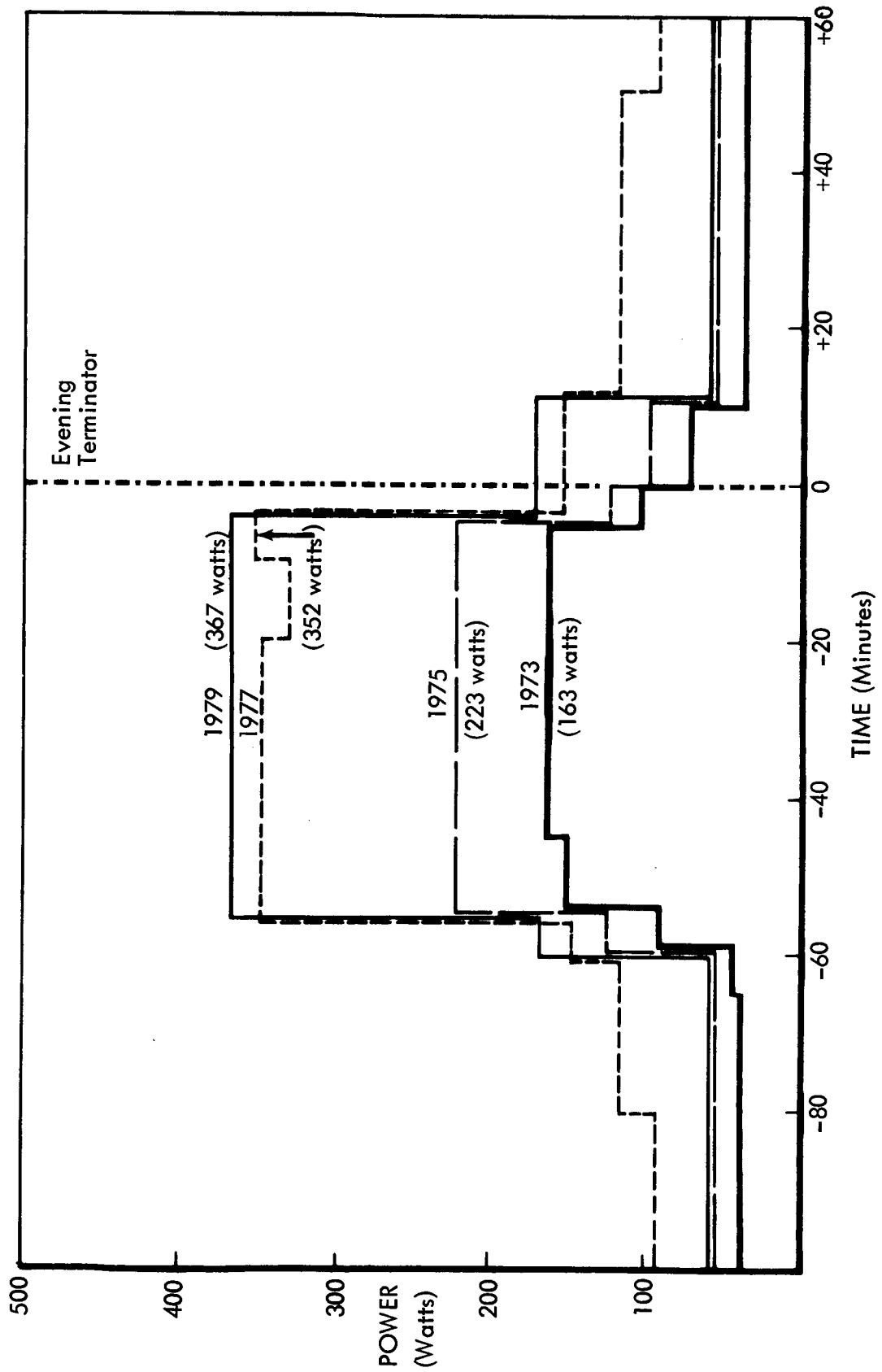


Figure 4-13: SCIENCE PAYLOAD POWER PROFILE EVOLUTION

Table 4-26: SCIENCE PAYLOAD DATA ACQUISITION RATES

Mission Experiments	1973		1975		1977		1979	
	Data Rate* (bits/sec)	Time (min)	Data Rate* (bits/sec)	Time (min)	Data Rate* (bits/sec)	Time (min)	Data Rate* (bits/sec)	Time (min)
Imaging	5×10^5	-24 to -5	5×10^5	-24 to -5	2.5×10^7	-24 to -5	2.5×10^7	-24 to -5
Broadband IR Spectrometer	1300/665	-29 to +10	1500	-29 to +10	1500	-29 to +10	1500	-29 to +10
Hi Resolution IR Spectrometer	1000	-29 to 0	4.6×10^4	-29 to +10	4.6×10^4	-29 to +10	4.6×10^4	-29 to +10
IR Radiometer	2300	-29 to +10	8000	-29 to 0				
UV Spectrometer ¹	4000	-29 to 0						
Plasma Probe	23	Continuous						
Cosmic Ray Telescope	46	Continuous	46	Continuous	1.6	Continuous		
Cosmic Dust Detector	1.6	Continuous						
Trap, Radiation Detector	15.5	Continuous						
Ion Chamber	15.5	Continuous	15.5	Continuous	600	-35 to +10		
Atmos. Polarimeter	300	-35 to +10						
Atmos. Mass Spectrometer	60	-15 to +10						
Subsatellite			500	Continuous	10	Continuous	500	Continuous
Bistatic Exper.			10	Continuous	4000	-50 to 20	10	Continuous
Bistatic Radar					4000	+20 to +50		Continuous
Solar Occultation			50	Continuous	200	Continuous	1	Continuous
Gamma Ray Spectrometer								
Neutron Albedo								
Meteor Flux					10	Continuous		
Celestial X-Ray					1500	Continuous		
Ionosphere Sounder			100	Continuous	100	Continuous		
Total B.P.S.	5.1×10^5		5.6×10^5		2.5×10^7		2.5×10^7	

¹ Time in minutes referred to evening terminator.² Alternate mode: UV spectrometer generates 2000 bits/sec from evening terminator to 60 min after evening terminator.

* Averaged over time of operation

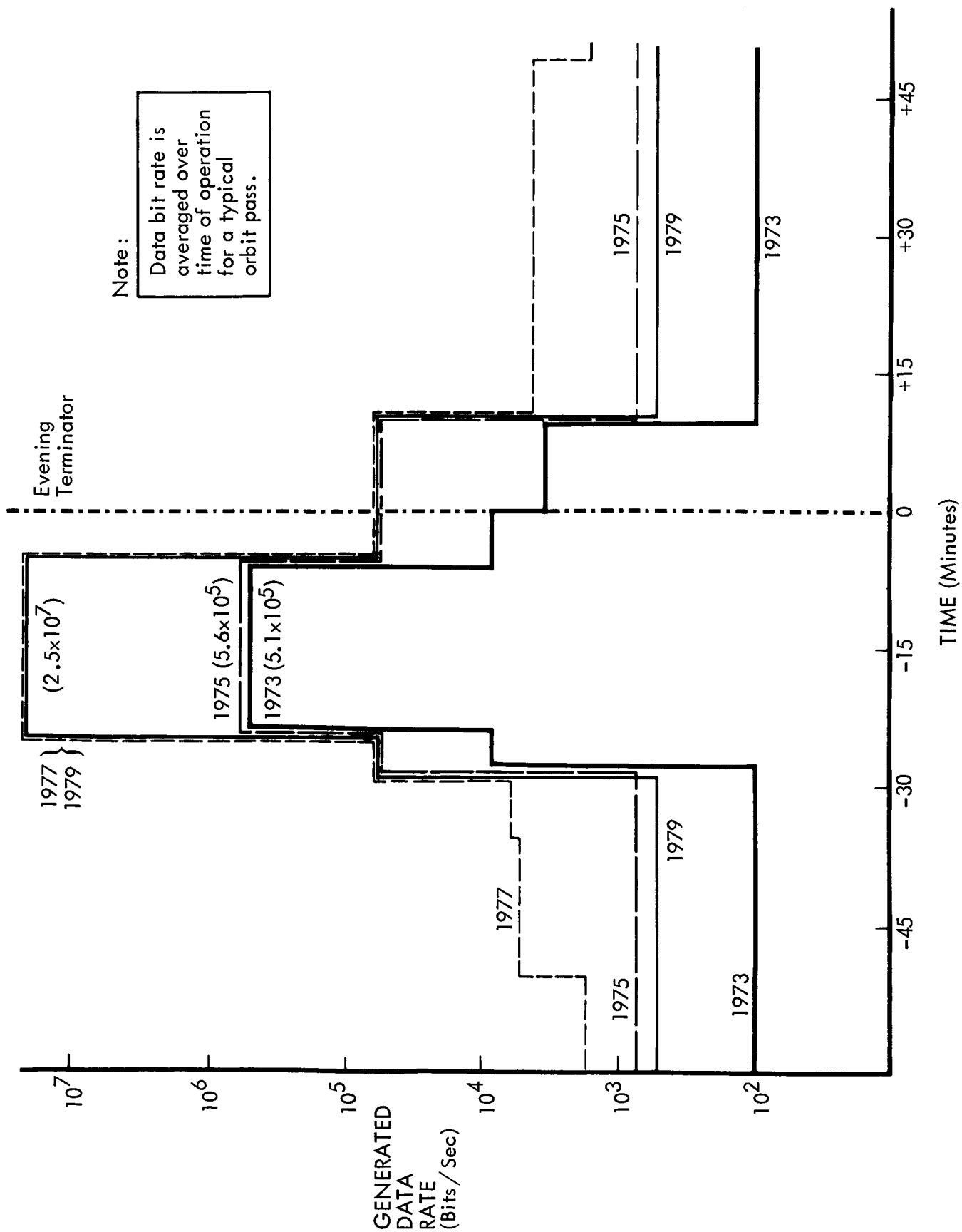


Figure 4-14: EVOLUTION OF SCIENCE GENERATED DATA PROFILE

4.3.8 Significant Changes to Spacecraft

The significant changes to the spacecraft as a result of expected evolution of experiments can be summarized in terms of weight, power, and data growth. The total weight of the orbital science payload grows from approximately 400 pounds in 1973 to 600 pounds in 1975, and to 1000 pounds in 1977 and 1979. As a result, the spacecraft launch weight in 1979 will increase by approximately 1500 pounds due to science payload evolution alone. The electrical power increases from about 160 watts in 1973 to about 370 watts for later missions. This increase in power can be accommodated by the 1973 configuration design through the addition of deployable solar panels and the addition of batteries. The growth of science data requires significant changes in the telecommunications and data storage subsystem. For advanced vidicon imaging systems in 1975, improved data recorders must be developed. For the film camera imaging system in 1977 and 1979, higher bit rate telecommunication systems must be developed. An RF analogue system with a larger antenna and power amplifier will be required. Alternatively, a laser communication system, if developed, could handle even larger amounts of data.

The large film camera in 1977 and 1979 will be body mounted. Spacecraft maneuvering will be required for picture taking, thereby increasing attitude control propellant requirements. Control moment gyros may also be required during picture taking to minimize smear and improve pointing.

4.3.9 Selected Science Payload Considerations

The following science payload related tasks were selected for further study:

- 1) Data automation equipment (DAE) centralization.
- 2) Computer simulation as a tool for science evaluation.

4.3.9.1 DAE Centralization

This study explored the advantages and disadvantages of combining the functions of the DAE with the computer and sequencer (C&S). The advantages are (1) equipment savings, (2) interface flexibility and simplicity, (3) increased command capability and flexibility, (4) improved data buffering control and flexibility, and (5) potential availability of limited data computing and processing. There appear to be no important technical disadvantages.

DAE functions, and their interfaces with the science instruments and C&S and data storage subsystem, are shown in Figure 4-15. The DAE command functions of timing, scan platform control, instrument sequencing, and instrument power switching are similar to those functions already performed in the C&S. These DAE functions could readily be performed in the C&S. The DAE functions associated with instrument output data multiplexing, formatting, and buffering are similar to functions performed by the data storage subsystem. These DAE functions could be included in the data storage subsystem.

Early spacecraft provided operational and scientific sequencing and computing functions within the hardware systems. Second generation spacecraft such as Mariner resulted in DAE, which is essentially a fixed wire sequencer with an override capability. The Lunar Orbiter added the reprogramming flexibility of a single address

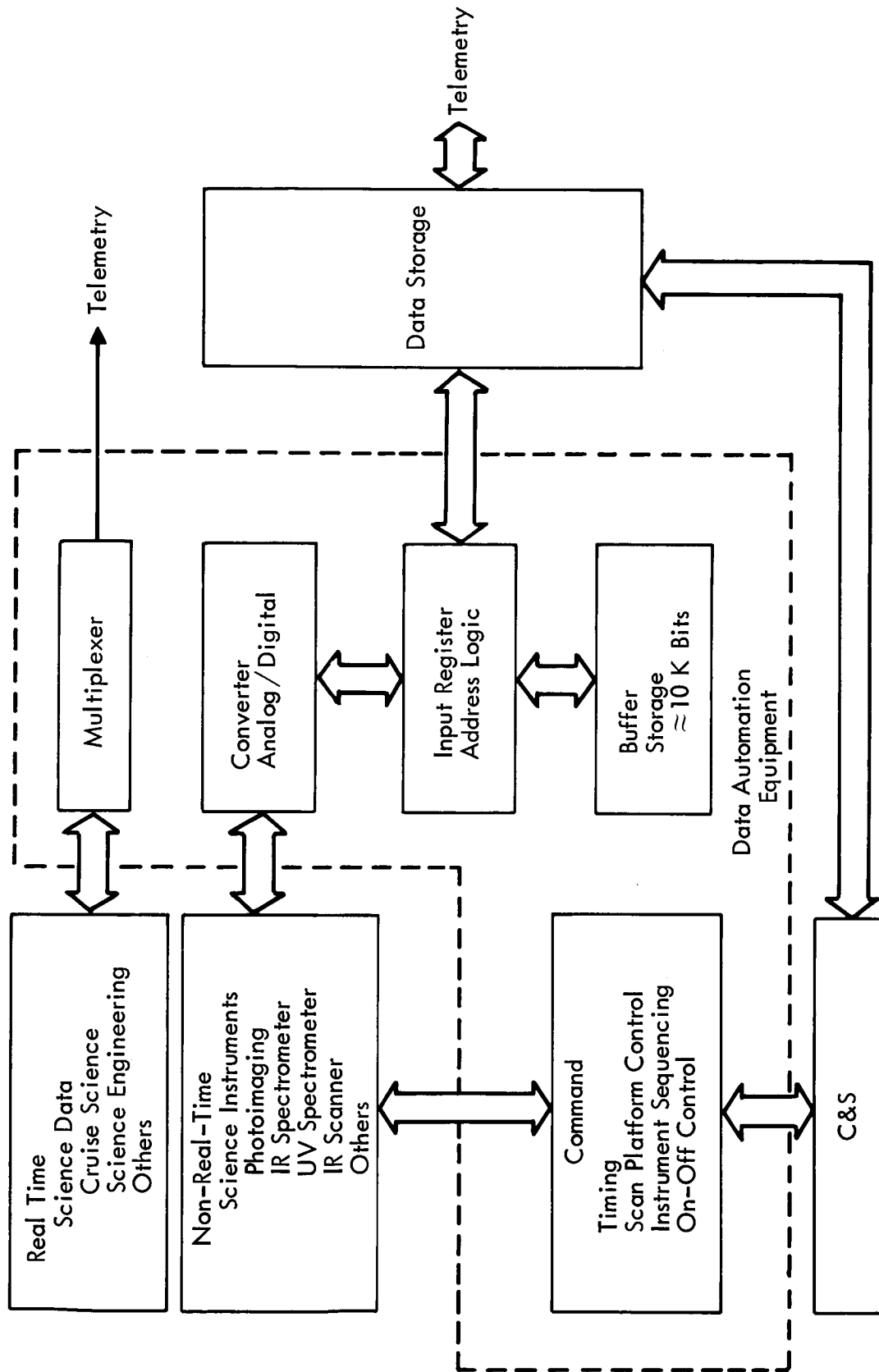


Figure 4-15: CENTRALIZED COMPUTER AND SEQUENCER

computer and included the computing requirements of maneuver angle integration for the attitude control subsystem. The Voyager C&S is a two-address computer fully capable of handling the DAE command function. The current design requirement is for 140 commands for all subsystems except the science subsystem. With its 1024 word capacity, the C&S can accept the additional loads of at least 150 science subsystem commands of the type shown in Table 4-27.

To take advantage of the flexibility of a reprogrammable computer, only a rudimentary total mission need be programmed at launch. This will keep computer memory size small. As various phases of the mission are completed, the computer will be reprogrammed. In this way, a larger portion of the computer memory capacity will be available to perform each mission phase as it occurs. The Lunar Orbiter missions turned out to be different from the design mission. The computer flexibility was instrumental in allowing the variations in mission operations without requiring hardware modification.

Figure 4-16 shows a typical DAE design. The bit, word, and frame counters, sequencers, and scan platform control and timing functions could be performed by the C&S. This would place the C&S interface at the science instruments. Commands necessary to address and format the data outputs would also be derived from the C&S. The total hardware required would be reduced. Only the driver circuitry to input the command signals to the instruments and output buffering equipment would need to be retained.

All of the above logical functions can be performed in the existing C&S. Since these logical functions are performed by computer programming, an additional benefit is obtained. Should it be necessary to interchange a given science package at some very late date, and if the required commands do not exceed the number available in that specific connector, then the problem is a simple one. Just a change in the instrument and its C&S control cable is required. Reprogramming can be accomplished at any time prior to actual instrument operation.

The buffer storage capability of the DAE represents a somewhat more complex problem. The inclusion of this buffering function in the data storage subsystem appears to be feasible. The command structure to control buffering and data storage would become more compatible with instrument operation. The inclusion of readout commands of the type listed in Table 4-27 would permit a control function over the address logic and input/output register as well as commutations. Tape control functions are also C&S-generated so that logical control element hardware could be reduced or eliminated.

The commutation, analog/digital conversion, addressing, and input/output register and buffer storage must be retained. This equipment runs at kilohertz and megahertz rates and cannot be accommodated in the C&S.

It is anticipated that with the freedom of control available under this arrangement, a savings in tape recorders could be achieved. The Task D design includes four high speed tape recorders ($3 \cdot 10^8$ bps) to match orbital readout at the high (48 K bps and 24 K bps) transmission rates. These recorders store mainly imaging data; however, all four will not be on simultaneously. Since the non-real-time data and the maneuver data is buffered, it may be possible to put this data on the high speed recorders when they are on standby. This possibly may eliminate low speed (10^6 and 10^7 bps) recorders with commensurate weight savings and should be the subject of further study.

Table 4-27: TYPICAL DATA AUTOMATION EQUIPMENT COMMANDS

<u>TV, CAMERA CONTROL</u> <ol style="list-style-type: none"> 1. On 2. Off 3. Focus Step 4. Shutter Setting 5. Shutter Setting 6. Filter Selection 7. Filter Selection 8. Mode Selection 9. Mode Selection 10. Gain Selection 11. Readout On 12. Readout Off 	<u>TV, MODE CONTROL</u> <ol style="list-style-type: none"> 1. TV Select (3 Cameras) 2. TV Select <u>SCAN PLATFORMS (TWO)</u> <ol style="list-style-type: none"> 1. Pitch + 2. Pitch - 3. Yaw + 4. Yaw - 5. Roll + 6. Roll - 7. Platform Select
<u>UV SPECTROMETER</u> <ol style="list-style-type: none"> 1. On 2. Off 3. Slit Step 4. Scan Rate 5. Scan Rate 6. Readout On 7. Readout Off 8. Cover Open 9. Cover Closed 10. Mode Select 11. Mode Select 12. Mode Select 13. Calibrate 	<u>HIGH RESOLUTION IR SPECTROMETER</u> <ol style="list-style-type: none"> 1. On 2. Off 3. Slit Step 4. Scan Rate 5. Scan Rate 6. Readout On 7. Readout Off 8. Cover Open 9. Cover Closed 10. Mode Select 11. Mode Select 12. Mode Select 13. Calibrate
<u>IR BROADBAND SPECTROMETER</u> <ol style="list-style-type: none"> 1. Science R/O 2. Engineering R/O 3. Channel Select 4. Channel Select 5. Science Multiplex 6. Engineering Multiplex 7. On 8. Off 9. Calibrate 	<u>IR RADIOMETER</u> <ol style="list-style-type: none"> 1. On 2. Off 3. Readout On 4. Readout Off 5. Calibrate 6. Gain Step

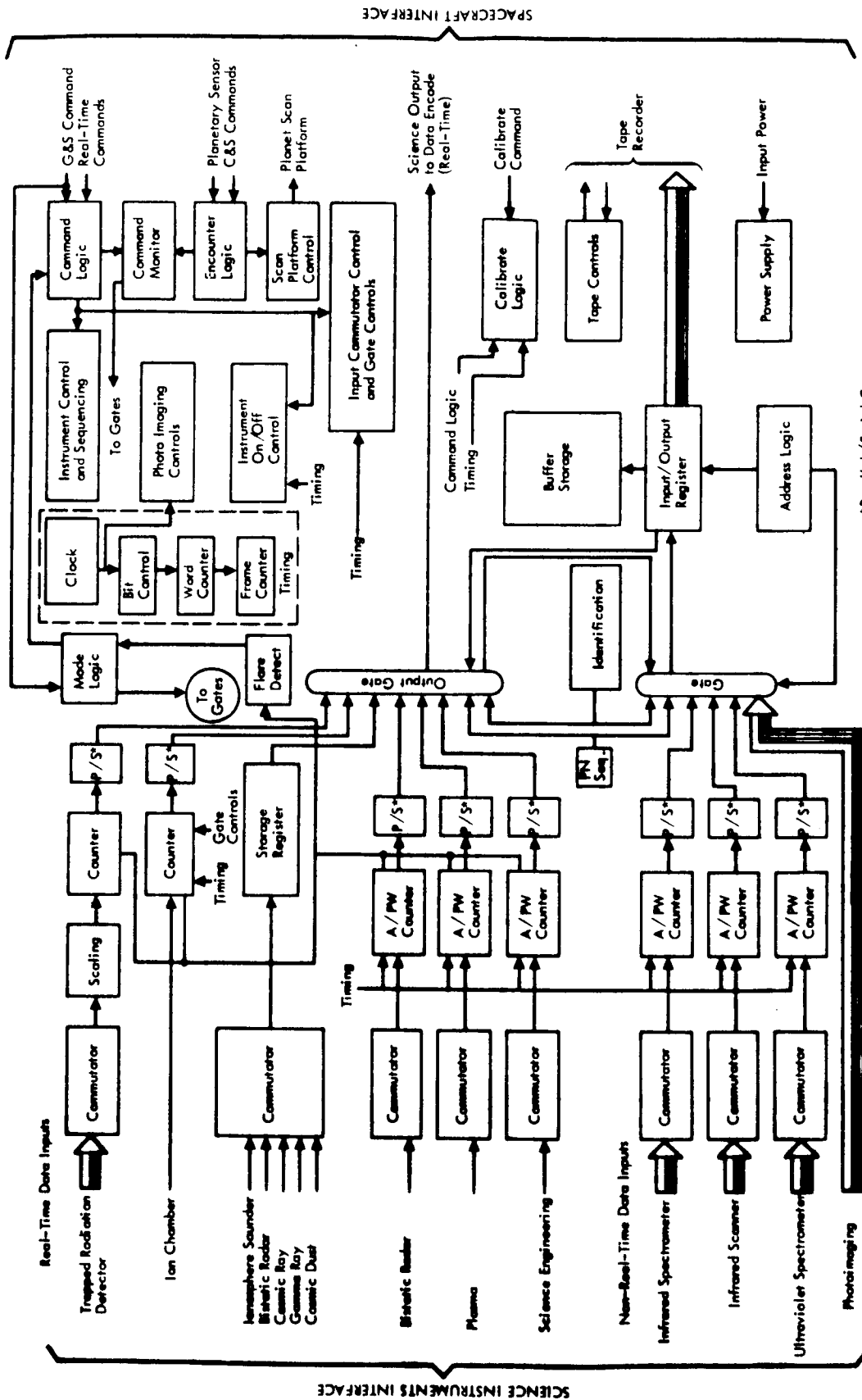


Figure 4-16: TYPICAL DATA AUTOMATION EQUIPMENT

Since direct contact would exist between the C&S and the data stream through the data control commands, the possibility of some limited data computing and processing in the C&S can be considered.

As a result of the foregoing it is concluded that the combining of the DAE command functions of timing, scan platform control, instrument sequencing, and power switching with those of the C&S is feasible. It is also concluded that the DAE functions associated with instrument output data multiplexing, formatting, and buffering can be combined with similar functions in the data storage subsystem.

4.3.9.2 Computer Simulation as a Tool for Science Evaluation

Digital and analog computer simulations are useful in design and analysis of spacecraft hardware and particularly of the science subsystem. The digital method lends itself to use early in the program to help establish subsystem definition. The development of timeline and event sequences early in the program is helpful in establishing power profiles, data profiles, scan platform motions, and instrument operational constraints. The analog method which generally evolves into model building, breadboarding, and combinations of hardware and analog computers, will undoubtedly be fully exercised during the program. Its use comes later in the program after subsystem definition and need not be discussed here.

Computer simulation programs can be of particular use in the early science subsystem analysis before design parameters are frozen. Mission simulation techniques will permit evaluating parameter changes. Sequencing and timelines based on the achievement of mission objectives can be generated. These will lead to power, data, orientation, command, and programming profiles.

The Lunar Orbiter program developed an event sequence and timeline computer program that permitted the generation of detailed mission events. The simulator permitted rapid analysis of the effects of parameter changes or perturbations. Typical perturbed elements were lighting angle, photography, processing, and video readout for different photo site locations.

A similar program would have high value in the operations analysis and design assessment of the Voyager spacecraft science subsystem. Such a program should include the following:

- 1) Detailed ephemeris data
- 2) Orbital parameters
- 3) Trans-Mars trajectory data
- 4) Science instrument constraints
- 5) Photo coverage data (number of frames, overlap, etc.)
- 6) Power
- 7) Instrument data rates, per sec, per orbit, per mission
- 8) Data rate constraints (i.e., recorder capacity, buffering, etc.)

- 9) Science instrument lighting angle requirements
- 10) Science instrument look angle
- 11) Science instrument clock and cone angles
- 12) Spacecraft coordinate systems
- 13) Times
 - Spacecraft
 - Terminator crossing
 - Sun eclipse
 - Earth occult
- 14) DSIF tracking data
- 15) DSIF constraints (lockup times, viewing periods, handovers, etc.)
- 16) Other spacecraft constraints (power, attitude control rates, maneuver times, computer and sequencer wait times, real-time command delay times).

Other parameters will undoubtedly be required as the program develops, but the above list indicates the large number of interrelating factors that control science subsystem operation. The program should be computerized at the earliest possible date for maximum utility. It must also be developed along guidelines that allow it to be adapted for operational use.

It is concluded that computer simulation would be a valuable tool for Voyager science subsystem analysis. Such a computer simulation program could be patterned after the Lunar Orbiter spacecraft mission event sequence and timeline computer program and use its existing software. Such a program should be initiated early in the preliminary design phase of the science subsystem so that it is functioning during Phase C.

4.4 CONCLUSIONS AND RECOMMENDATIONS

As a result of this study task the following conclusions are made:

- 1) Evolution of the photoimaging experiment causes the greatest impact to the spacecraft. Its growth from 150 pounds in 1973 to 660 pounds in 1977 and 1979 is required to support the basic science objectives relating to the origin and evolution of the solar system and of extraterrestrial life. Photoimaging is the most important single orbiter experiment in accomplishing these basic science objectives.
- 2) The spacecraft orbiter best lends itself to remote electromagnetic sensing over broad areas and throughout seasonal variations. In addition to remote sensing in the visible spectrum (photoimaging), spectral measurements in the infrared and ultraviolet regions will contribute heavily to meeting the basic science objectives. Evolution of spectral instruments will be in the direction of increased spectral and spatial resolution and in broader spectral range. These changes will result in increased weight. This impact on the spacecraft will be considerably less than that caused by photoimaging evolution.

- 3) The subsatellite experiment (orbital experiment capsule) provides the capability for obtaining measurements of the Mars magnetic field as low as 0.25 gamma away from the magnetic influence of the spacecraft. In addition, it provides a convenient measurement base for electromagnetic probing of the Mars atmosphere, using a bistatic link between the subsatellite and the spacecraft.
- 4) Evolution of the total science payload results in a weight increase from approximately 400 pounds in 1973 to 600 pounds in 1975 and 1000 pounds in 1977 and 1979. Increases also are experienced in volume, power, and data. In spite of these significant changes to the payload, it is expected that their impact on the spacecraft can be accommodated by the 1973 configuration design.
- 5) The future direction of science experiment evolution will be dependent on planetary measurement results. If data returned from early spacecraft indicate the possibility of life, then experiment emphasis will shift from orbiter to lander. Similarly, as data accumulate showing interesting science areas or phenomena, emphasis will shift to different experiments in the spacecraft. For this reason, and to take care of prelaunch contingencies, every effort should be taken to provide as much flexibility as possible in the science payload, e.g., modularization of science experiment electronic packages.
- 6) A major problem area concerns the development of a higher bit rate telecommunication system (and possibly an improved data storage system) to handle the much larger amount of scientific data that will be generated in subsequent missions.
- 7) The centralization of data automation equipment (DAE) functions in the computer and sequencer (C&S) and data storage subsystem results in the following advantages: 1) equipment savings; 2) interface flexibility and simplicity; 3) increased command capability and flexibility; and 4) improved data buffering control. This centralization would be accomplished by 1) combining the DAE command functions of timing, scan platform control, instrument sequencing, and power switching within the C&S; and 2) combining the DAE functions associated with instrument output data multiplexing, formatting; and buffering with similar functions in the telecommunication subsystem (data storage).

The following recommendations are suggested for further work related to the science subsystem:

- 1) Development of higher bit rate telecommunication systems, such as laser, leading to capabilities of from 300,000 to 1,000,000 bits per second.
- 2) Better definition of experiments and their characteristics to permit clarification of spacecraft and interface requirements.
- 3) Further investigation of DAE functions and the determination of optimum interface separations between the science instruments, DAE, C&S, and data storage subsystem.
- 4) Development of a computer simulation program as a tool in science subsystem analysis.

- 5) Further investigation of all science function divisions to determine the optimum interface separations between the GFE science packages and the integrated spacecraft. Such science support equipment as scan platforms, booms, DAE, power switching electronics, and cabling are probably more susceptible to effective integration by the spacecraft contractor than the scientific principal investigators.

5.0 PARTICULATE CONTAMINATION CONSIDERATIONS

5.1 OBJECTIVES

The objectives of this task were to (1) determine the most reasonable level of particulate contamination (nonbiological) control for the Voyager flight spacecraft and its subsystems, and (2) show the impact of that level of control on required facilities, including costs.

5.2 SCOPE

Particulate contamination must be controlled during spacecraft processing and assembly, as it can influence spacecraft operational reliability. For example, evidence from the flight of Mariner IV indicates that interference with the Canopus sensor and consequent loss of lock was caused by loose fragments that caused confusing reflections to be seen by the optical sensors.

Particulate contamination also is of concern to pneumatic systems (e.g., the reaction control and propulsion subsystems) where close tolerance valves are used. In electronic circuits, contamination can cause short circuits and circuit characteristics changes. Thermal control coatings also can be affected by particulate contamination where emissivity and absorptivity are rigidly controlled.

Particulate contamination is controlled by fabricating, assembling, and processing in clean rooms. Costs of clean room facilities and operations can be high for Voyager-size spacecraft. Consequently, the maximum allowable level of contamination and the attendant choice of clean room required must be determined realistically.

5.3 STUDY APPROACH

The approach used in this study is illustrated in Figure 5-1. Each of the tasks accomplished in this study, as described below, is keyed to the numbers on the figure.

Task 1 -- Existing data on industrial and government agency clean rooms were reviewed. Information on capabilities for controlling various levels and types of contamination, cost of facilities, and limitations and advantages were collected.

Task 2 -- Applicable data collected in Task 1 were analyzed to identify existing techniques and methods for Voyager.

Task 3 -- This task consisted of a review of the latest design of the Voyager spacecraft subsystems to determine the criticality of each subsystem or component to particulate contamination. This task was performed concurrent with Tasks 1 and 2.

Task 4 -- From the subsystem criticality analysis, allowable limits of size and type of contamination levels were determined.

Task 5 -- Based on the results of Tasks 2 and 4, candidate techniques were identified for attaining allowable particulate contamination limits for each spacecraft subsystem.

Task 6 -- From Tasks 4 and 5, the most reasonable technique for controlling and monitoring of the contamination level, and the facilities required, were determined.

Task 7 -- Cost data were obtained based on Lunar Orbiter manufacturing and processing operations.

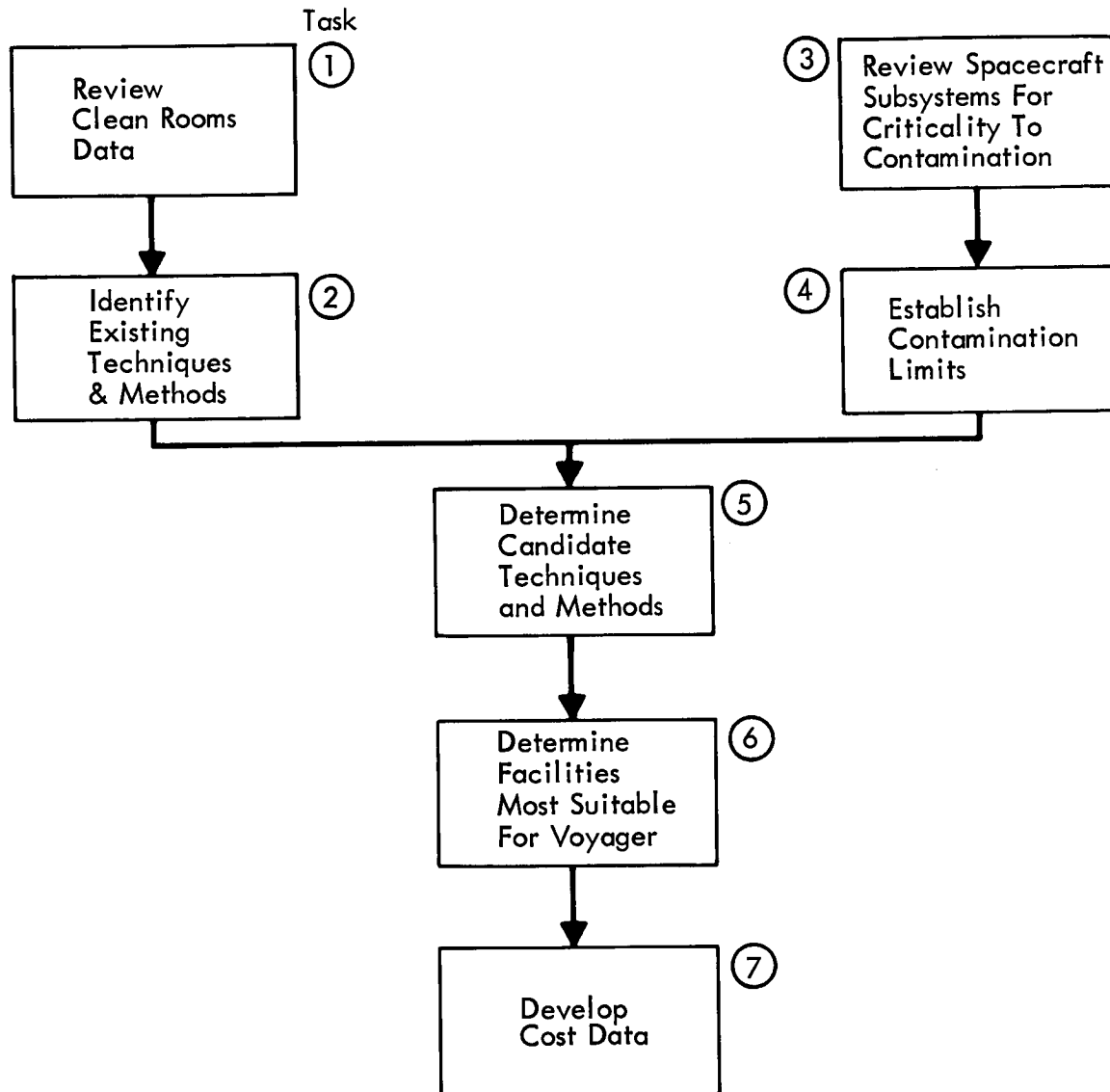


Figure 5-1: PARTICULATE CONTAMINATION STUDY PLAN

Study results are summarized in the following section.

5.4 RESULTS

The results of this study will be discussed in three sections, as follows: (1) Subsystem Requirements, (2) Cleaning Processes and Clean Room Facilities, and (3) Cost Data.

5.4.1 Subsystem Requirements

Subsystem requirement identification was initiated by reviewing the Lunar Orbiter contamination control requirements. New developments in the monitoring and control of clean environments and clean room facilities were then investigated. As a result, the following data pertaining to subsystem cleanliness requirements were obtained:

- Maximum size and number and type of allowable contamination
- Critical hardware
- Contamination impact
- Particulate contamination controls for various levels of testing.

These data are summarized in Table 5-1. For each identified critical subsystem the following information is provided in the table:

- 1) Allowable contamination limit for each type of operation.
- 2) Proposed technique and equipment required to meet allowable contamination limits.
- 3) Type of facility required.

In addition to the tabulated requirements, the following procedures apply:

- 1) White nylon gloves shall be worn when thermal control surfaces are handled. Gloves shall be changed periodically (3 to 4 hours).
- 2) No cables and wires, especially those which may have grease, oils, or loose contaminants, shall contact the thermal control finishes.
- 3) Oil-diffusion-pumped vacuum chambers shall not be used for testing the spacecraft or any component.
- 4) Spacecraft coolant gas shall be controlled to allow no particles greater than 100 microns in size, less than 700 particles of 5 microns and larger, and less than 100,000 particles 0.5 microns or larger per cubic foot. In addition, hydrocarbons are limited to a maximum of 3 parts per million. Temperature and dew point shall be controlled to eliminate condensation.
- 5) The assembled solar array may be exposed to normal shop environments, provided that the surfaces are cleaned prior to flight. Plastic covers are recommended for protective use during nonoperating periods.

Table 5-1: PARTICULATE CONTAMINATION REQUIREMENTS AND FACILITIES
(SHEET 1 OF 3)

CRITICAL SUBSYSTEM	TYPE OF OPERATION	CONTAMINATION LIMITS	PROPOSED TECHNIQUE & EQUIPMENT REQUIRED	TYPE OF FACILITY REQUIRED	REMARKS & PRECAUTIONS
PROPULSION SUBSYSTEM	1. SUBASSEMBLY OF PARTS & COMPONENTS	NONE ABOVE: 5 μ METALLIC 25 μ NONMETALLIC (INTERNAL SURFACES)	TECHNIQUES AND EQUIPMENT SIMILAR TO THOSE DESCRIBED IN LUNAR ORBITER D2-100411-1. CLEAN AS ASSEMBLE. MAINTAIN CLEANLINESS LEVELS.	CLASS 100: CLEAN BENCHES OR BOOTH (DOWNFLOW). POSSIBLE USE OF HORIZONTAL FLOW ROOM.	WILL REQUIRE TRAINED PERSONNEL
	2. ASSEMBLY INTO SUBSYSTEMS				
	3. SUBSYSTEM QUAL. TESTS				
	4. SPACECRAFT ASSEMBLY				
	5. SUBSYSTEM INTEGRATION TESTS				
	6. SPACECRAFT ACCEPTANCE TESTS				
	7. SPACECRAFT CHECKOUT & LAUNCH OPERATIONS	\triangle $< 5 \mu$ NO LIMITS 5-25 μ < 220 28-100 μ < 10 NONE ABOVE 100 μ		CLASS 100,000	
POWER SUBSYSTEM - SOLAR ARRAY ASSEMBLY	1. SUBASSEMBLY OF PARTS & COMPONENTS	\triangle 10 μ < 700 PER 1 in. ² PER 8 HR (BETWEEN CELLS & COVER GLASS) NORMAL GOOD HOUSEKEEPING STANDARDS \triangle 0-5 μ & LARGER $< 100,000$ 5 μ & LARGER < 70 NONE ABOVE 100 μ	CLASS M BAC 5703 NO CONTROL CLEAN AS ASSEMBLE MAINTAIN CLEANLINESS LEVELS	CLASS 100,000 ——— CLASS 100,000	OPERATION NO. 1- ENVIRONMENTAL CONTROLS BEING CALLED OUT ARE LESS STRINGENT THAN THOSE FOR STEPS NO. 4, 5, 6 & 7. REQUIRES EXTENSIVE USE OF TRAINED PERSONNEL.
	2. ASSEMBLY INTO SUBSYSTEMS				
	3. SUBASSEMBLY QUAL. TESTS				
	4. SPACECRAFT ASSEMBLY				
	5. SUBSYSTEM INTEGRATION TESTS				
	6. SPACECRAFT ACCEPTANCE TESTS				
	7. SPACECRAFT CHECKOUT & LAUNCH OPERATIONS				

\triangle Maximum Number of Particles Per Square Foot of Significant Surface

Table 5-1: PARTICULATE CONTAMINATION REQUIREMENTS AND FACILITIES
(SHEET 2 OF 3)

CRITICAL SUBSYSTEM	TYPE OF OPERATION	CONTAMINATION LIMITS	PROPOSED TECHNIQUE & EQUIPMENT REQUIRED	TYPE OF FACILITY REQUIRED	REMARKS & PRECAUTIONS
GUIDANCE & CONTROL SUBSYSTEM - REACTION CONTROL ASSEMBLY	1. SUBASSEMBLY OF PARTS & COMPONENTS	NONE ABOVE: 5 μ METALLIC 25 μ NONMETALLIC (INTERNAL SURFACES)	TECHNIQUES AND EQUIPMENT SIMILAR TO THOSE DESCRIBED IN LUNAR ORBITER D2-100411-1 AND THE PROPULSION SYSTEM LISTED ABOVE.	CLASS 100: CLEAN BENCHES OR CLEAN BOOTHS OR MODIFICATION OF SAME. SAME COMMENTS AS PROPULSION SUBSYSTEM.	TRAINED PERSONNEL REQUIRED.
	2. ASSEMBLY INTO SUBSYSTEMS				
	3. SUBASSEMBLY QUAL. TESTS				
	4. SPACECRAFT ASSEMBLY				
	5. SUBSYSTEM INTEGRATION TESTS				
	6. SPACECRAFT ACCEPTANCE TESTS				
	7. SPACECRAFT CHECKOUT & LAUNCH OPERATIONS	< 5 μ NO LIMIT Δ 5 - 25 μ < 200 25 - 100 μ < 10 NONE ABOVE 100 μ	CLASS 100,000		
GUIDANCE AND CONTROL SUBSYSTEM - ELECTRONICS AND SENSORS	1. SUBASSEMBLY OF PARTS & COMPONENTS	NONE ABOVE: 5 μ METALLIC 25 μ NONMETALLIC (INTERNAL)	TECHNIQUES AND EQUIPMENT SIMILAR TO THOSE DESCRIBED IN LUNAR ORBITER DOCUMENTATION	CLASS 100	WILL REQUIRE SPECIALIZED TRAINING CLASSES FOR WORKING PERSONNEL. USE TECHNIQUES SIMILAR TO PROCESS IN LUNAR ORBITER DOCUMENTS D2-100287-1, BAC 5123, AND BAC 5703.
	2. ASSEMBLY INTO SUBSYSTEMS	NORMAL GOOD HOUSEKEEPING STANDARDS	NO CONTROL	_____	
	3. SUBSYSTEM QUAL. TESTS		CLEAN AS ASSEMBLE	CLASS 100,000	
	4. SPACECRAFT ASSEMBLY		MAINTAIN CLEANLINESS LEVELS		
	5. SUBSYSTEM INTEGRATION TESTS				
	6. SPACECRAFT ACCEPTANCE TESTS				
	7. SPACECRAFT CHECKOUT & LAUNCH OPERATIONS.	0.5 μ & LARGER < 100,000 5 μ & LARGER < 700 NONE ABOVE 100 μ			

Δ Maximum Number of Particles Per Square Foot of Significant Surface

Table 5-1: PARTICULATE CONTAMINATION REQUIREMENTS AND FACILITIES
(SHEET 3 OF 3)

CRITICAL SUBSYSTEM	TYPE OF OPERATION	CONTAMINATION LIMITS	PROPOSED TECHNIQUE & EQUIPMENT REQUIRED	TYPE OF FACILITY REQUIRED	REMARKS & PRECAUTIONS
SCIENCE SUBSYSTEM - SCIENCE SENSORS	1. SUBASSEMBLY OF PARTS & COMPONENTS	0.5 μ & LARGER \triangle < 100 1 μ & LARGER < 10 NONE ABOVE 4 μ	SAME AS GUIDANCE & CONTROL SUBSYSTEM	CLASS 100	SAME AS GUIDANCE & CONTROL SUBSYSTEM ELECTRONIC ASSEMBLY
	2. ASSEMBLY INTO SUBSYSTEMS	NORMAL GOOD HOUSEKEEPING STANDARDS		CLASS 100,000	
	3. SUBSYSTEM QUAL. TESTS				
	4. SPACECRAFT ASSEMBLY				
	5. SUBSYSTEM INTEGRATION				
	6. SPACECRAFT ACCEPTANCE TESTS				
	7. SPACECRAFT CHECKOUT & LAUNCH OPERATIONS				
SCIENCE SUBSYSTEM - SCAN PLATFORMS	1. SUBASSEMBLY OF PARTS & COMPONENTS	NONE ABOVE: 5 μ METALLIC 25 μ NON-METALLIC (INTERNAL SURFACES)	SAME AS GUIDANCE & CONTROL SUBSYSTEM	CLASS 100	SAME AS GUIDANCE & CONTROL SUBSYSTEM ELECTRONIC ASSEMBLY
	2. ASSEMBLY INTO SUBSYSTEMS	NORMAL GOOD HOUSEKEEPING STANDARDS		CLASS 100,000	
	3. SUBASSEMBLY QUAL. TESTS				
	4. SPACECRAFT ASSEMBLY				
	5. SUBSYSTEM INTEGRATION				
	6. SPACECRAFT ACCEPTANCE TESTS				
	7. SPACECRAFT CHECKOUT & LAUNCH OPERATIONS				
STRUCTURAL & MECHANICAL SUBSYSTEM - DEPLOYMENT MECHANISMS	1. SUBASSEMBLY OF PARTS & COMPONENTS	NONE ABOVE: 5 μ METALLIC 25 μ NON-METALLIC (INTERNAL SURFACES)	TECHNIQUES & EQUIPMENT SIMILAR TO THOSE DESCRIBED IN LUNAR ORBITER DOCUMENTATION	CLASS 100	USE TECHNIQUES SIMILAR TO PROCESSES IN LUNAR ORBITER DOCUMENTS: D2-100290-1, D2-100352-1, BAC 5956, D2-100301-1, D2-100273-1, BAC 5750, & D2-100358.
	2. ASSEMBLY INTO SUBSYSTEMS	NORMAL GOOD HOUSEKEEPING STANDARDS	NO CONTROL	CLASS 100,000	
	3. SUBASSEMBLY QUAL TESTS				
	4. SPACECRAFT ASSEMBLY				
	5. SUBSYSTEM INTEGRATION				
	6. SPACECRAFT ACCEPTANCE TESTS				
	7. SPACECRAFT CHECKOUT & LAUNCH OPERATIONS				

\triangle Maximum Number of Particles Per Square Foot of Significant Surface

- 6) Optical surfaces and lenses shall be cleaned prior to spacecraft encapsulation.
- 7) The spacecraft shall be assembled and tested in an environment that limits the particulate contamination to 100 microns or less in size.

5.4.2 Cleaning Processes and Clean Room Facilities

This section discusses the following:

- 1) Cleaning Processes
- 2) Clean Room Considerations
- 3) Spacecraft Level Cleanliness Verification
- 4) Clean Room and Handling Procedures

Cleaning processes, procedures, and facilities for the Voyager spacecraft will be similar to those used for the Lunar Orbiter program. Representative procedures used on the Lunar Orbiter program are listed in Table 5-2.

5.4.2.1 Cleaning Processes

There are two types of cleaning that must be considered. These are gross cleaning and final cleaning. The latter type is sometimes referred to as supercleaning.

Requirements for gross cleaning are contained in existing Boeing specifications. Final cleaning requirements are not available and must be developed. Final cleaning is usually accomplished by use of solvents or detergent solutions. In either case, the objective is to dissolve the contaminant, if soluble, and to suspend and flush away insoluble materials. The solution or solvent and the process used will depend on such considerations as the material, configuration, cleanliness level, kind of soil, and economy. Process control is a major factor in ensuring that the desired cleanliness is achieved and maintained. In general, the process will consist of one or more of the following:

- Hot or cold cleaning
- Vapor degreasing
- Spray cleaning
- Ultrasonic cleaning
- Flush cleaning.

All process fluids require filtration to remove particulate contamination.

Table 5-2: TYPICAL LUNAR ORBITER FABRICATION AND CLEANING PROCEDURES

HARDWARE APPLICATION	CLEANLINESS REQUIREMENT	BOEING DOCUMENTATION
Reaction Control & Propulsion - cleaning	No Metallic particles Above 5 Microns No Nonmetallic Particles Above 25 Microns	D2-100411-1
Wire Harness - fabrication & installation	Airborne Particles: Up To 1500 /ft ³ None Above 10 Microns Airborne Hydrocarbons: 3 ppm max	D2-100287-1
Thermal Coating - application	Clean Spray Booth Water Break Free Part Surface	D2-100290-1
Thermal Barrier - construction	Perform in Class 100,000 Clean Room, Except Spray Coating of Dacron	D2-100352-1
Induction Brazed Tubing - fabrication & repair	Same as Reaction Control and Propulsion Repaired Parts - Clean to Initial Level Before Reinstallation	D2-100301-1
Nitrogen Tank - fabrication	Acetone Flush Etch Clean	D2-100273-1
Preparation of Hardware for Assembly in Class 100,000 Clean Room	Solvent Clean Mechanical Clean (vacuum or blower) Package in Aclar Bag	D2-100358-1
Ground Servicing & Checkout Equipment - cleaning	Particle Limit per Square Foot of Significant Surface	D2-100465-1

5.4.2.2 Clean Room Considerations

A clean room is a special facility for maintaining a required cleanliness level. Voyager particulate contamination requirements can be met by Class 100,000 clean rooms and Class 100 areas within the Class 100,000 clean room. These two facilities per Federal Standard No. 209a, are characterized by the particle size/density distribution curves shown in Figure 5-2.

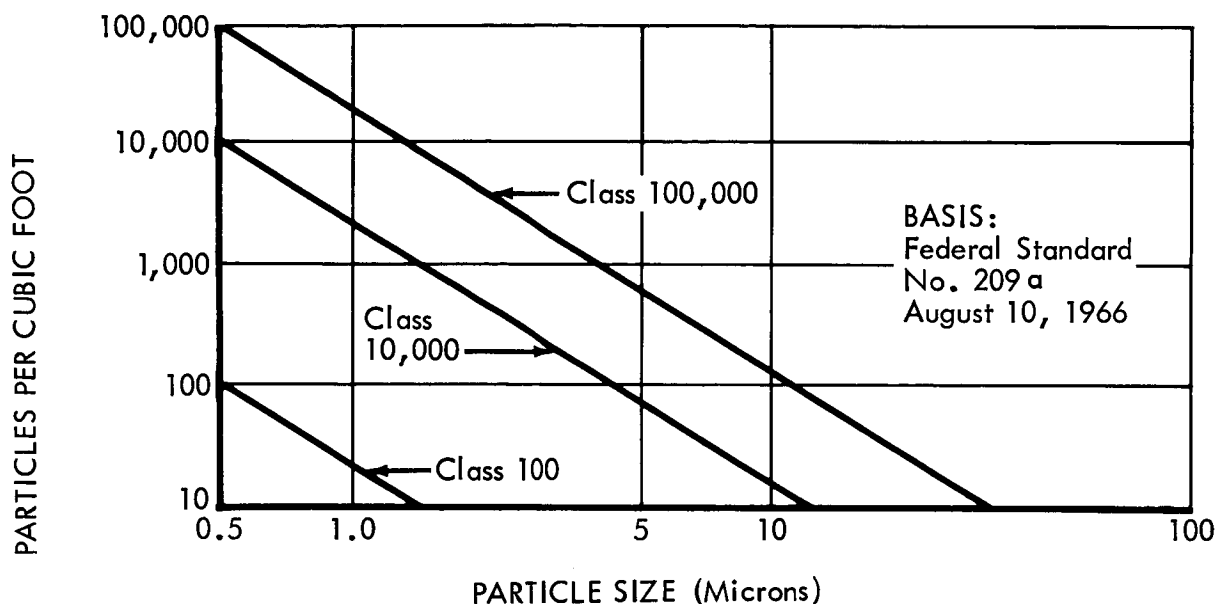


Figure 5-2: CLEAN ROOM CHARACTERISTICS

The unique characteristics of a Class 100 clean area are summarized below.

Class 100 Per Federal Standard No. 209a (Particles per Cubic Foot)

0.5 μ and larger < 100

1.0 μ and larger < 20

4.0 μ and larger < 1

Class 100 is achieved on a practical basis with clean benches, operating immediately next to a high-efficiency-particulate-air (HEPA) filter bank in a Class 100,000 laminar flow clean room. Only tools designed to minimize particle generation are permitted. Personnel must always remain downstream of the hardware. The unique requirements of a Class 100,000 clean room are summarized below.

Class 100,000 Per Federal Standard No. 209a (Particles per Cubic Foot)

0.5 μ	and larger	<	100,000
5.0 μ	and larger	<	700
25.0 μ	and larger	<	20

Major modification of existing structures or construction of entirely new facilities will be required to accommodate Voyager-size payloads.

5.4.2.3 Spacecraft Level Cleanliness Verification

A direct verification of the flight spacecraft cleanliness level is not feasible. Strict compliance with processing and contamination control procedures is required to ensure that the required spacecraft level cleanliness is achieved. To a limited extent, visual verification of cleanliness is feasible.

5.4.2.4 Clean Room and Handling Procedures

Operation of a clean room facility is the controlling factor in maintaining a cleanliness level. Particular attention must be given to procedures for the following:

- Personnel cleaning prior to entry
- Control of personnel head count
- Cleaning material prior to entrance
- Control of special items needed
- Traffic control
- Layout of area
- Exposure time of hardware
- Clean room garments
- Maintenance requirements
- Environment monitoring
- Tools, tool cleaning, and calibrating
- Continuous cleaning as assembly occurs
- Visual inspection
- Trained personnel

- Personnel motivation
- Location of work relative to filter bank.

The following defines general procedures and environment required for fabricating, assembling, and testing the Voyager spacecraft. For discussion purposes, spacecraft subsystems are divided into three categories: (1) computing and sequencing, telecommunications, cabling, power, and pyrotechnics; (2) propulsion and attitude control; and (3) structural and mechanical (including thermal control). In group (1), individual components will be received from point of manufacture in a cleaned double-bagged condition. Bag opening for test and inspection will be accomplished in a Class 100,000 clean room. Individual operations that require a higher degree of cleanliness will be accomplished in a Class 100 clean area. Structural components (containers for electronic subsystems) will be fabricated in normal shop environments and then cleaned, painted, and moved into a controlled cleaning facility where the part will be given a final cleaning and packaged. If it becomes necessary to remove the part from the package, it will be done in a Class 100,000 clean room. Should it be necessary to remove the part from the clean room for an unplanned operation in a noncontrolled environment, the part will go back through the cleaning and packaging operation before being readmitted to the clean room. Soldering and welding of electronic parts will be accomplished within the areas that meet or exceed the requirements established by NASA Quality Publication NPC 200-4, August 1964, Section No. 2. Housekeeping standards and procedures established for Lunar Orbiter will be used as a baseline for developing Voyager standards and procedures. The knowledge gained on Lunar Orbiter operations indicates that frequent cleaning, wiping, and vacuuming of parts is necessary to maintain a high level of parts cleanliness.

Group (2) items will be treated as group (1) items except where tubing, valves, regulators, and other plumbing type items are involved. The internal cleanliness is achieved by a cleaning and flushing operation performed within an area meeting Class 100 particulate limitations. Cleanliness levels of interior surfaces are verified by measuring the contaminants deposited by the flushing solution on a millipore filter. The cleaning, flushing, checking, and packaging operation will be accomplished within a Class 100 environment without leaving the clean bench. The assembly (brazing) of these supercleaned parts will take place in a Class 100,000 clean room, within a Class 100 clean bench. All fluids used in subsequent testing will be cleaned to meet the required level of cleanliness. Once the components have been assembled, sealed, and if necessary pressurized, they will not be opened except in an appropriate environment. The exterior of the systems can be exposed to noncontrolled environments without being detrimental to the subsequent operation of the system. However, external cleaning procedures (wiping and vacuum) will be used after such exposure and before admittance to a controlled area.

Group (3) items will be fabricated and preassembled in normal factory areas using the normal, high quality, housekeeping standards. When the prefit operation has been completed, the parts will be cleaned, painted, and then recleaned and packaged for clean room assembly. After the subsystems have been assembled into the spacecraft, the assembled spacecraft will be moved between controlled locations only after the spacecraft has been properly sealed with a protective barrier.

Once the flight spacecraft have completed their Seattle testing and are transported to Cape Kennedy in a previously cleaned and sealed container, they should not be exposed to an environment containing particulate matter that exceeds a Class 100,000 limit.

5.4.3 Cost Data

The following discussion provides cost data on clean room facilities and clean room operations.

5.4.3.1 Facilities

The cost data for clean room facilities construction provided herein are based on Seattle area costs. Construction costs for a Class 100,000 clean room were found to vary from a minimum of \$40/square foot for a horizontal laminar flow room to a maximum \$120/square foot for a vertical laminar down-flow room. Both figures assume that the clean room is built within an existing building. Much of the difference in costs is dependent upon existing building condition, clearance height, availability of utilities, etc., and the extent of existing support areas (e.g., personnel change area, air shower, intercom, and monitoring techniques). The construction costs for a given type of construction for a Class 100,000 clean room are the same as for a Class 10,000 or 100 clean room. The difference in cleanliness levels is obtained by the functional operation performed within the clean room. For example, a Class 100,000 room may accommodate 20 people within the area, performing moderately strenuous activities. A Class 10,000 environment within the same room may be achieved by reducing the number of people, restricting the activities performed, and cleaning the items being worked on more frequently. To achieve a Class 100 environment, it would be necessary to impose even more stringent operating restrictions. Figure 5-3 shows clean room unit costs versus ceiling height for three types of clean rooms. The wall-to-wall laminar flow clean room is the least expensive and is preferred for Voyager.

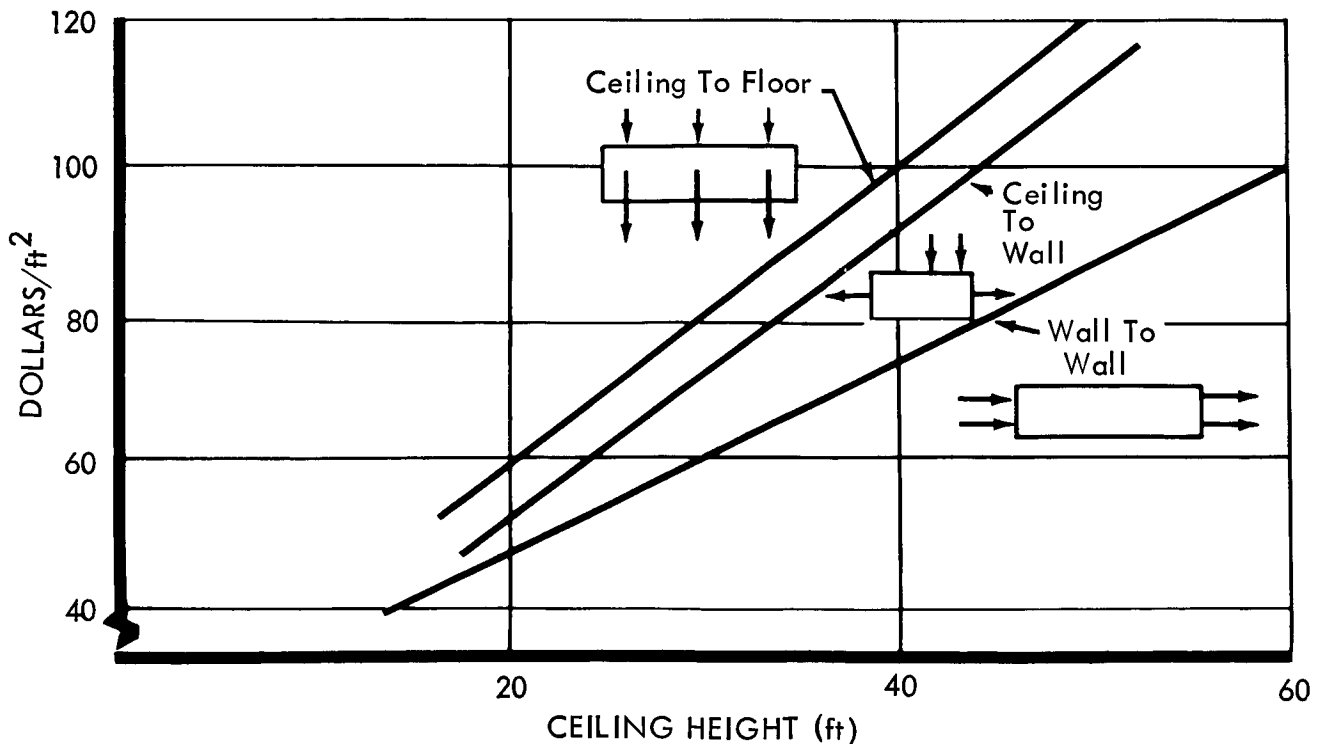


Figure 5-3: CLEAN ROOM FACILITY UNIT COST

5.4.3.2 Operations

Incremental costs associated with clean room operations were developed. Data accumulated on the Lunar Orbiter program were used. The following table presents typical data for a Lunar Orbiter plumbing assembly:

<u>Operation</u>	<u>Normal Manufacturing Time (Manhours)</u>	<u>Manufacturing Time (Including Cleaning) (Manhours)</u>
Fabrication	518	518
Test (Operation)	133	133
Assemble & Weld	187	187
Fit-up	284	284
Disassemble	---	55
Clean	---	149
Test (Cleanliness)	---	266
	<hr/>	<hr/>
Totals	1,122	1,592

As noted above, the time required to satisfy part cleanliness for this specific system was 470 manhours or 42% increase. Based on Lunar Orbiter experience, an overall increase of 14% in manhours has been estimated for the cleaning operation of the complete vehicle based on a weighted average of the following:

<u>Operation</u>	<u>Additional Manhours for Cleaning</u>
Make Parts	10%
Plumbing	42%
Electrical and Electronic	10%
Assembly	12.5%

5.5 CONCLUSIONS

The following conclusions are reached:

- 1) Clean room facilities are required for Voyager.
- 2) The most reasonable level of Voyager particulate contamination control requires a Class 100,000 room. Limited operations on a Class 100 bench within a Class 100,000 room also are required.

- 3) A wall-to-wall laminar flow clean room is preferred because of its relatively low cost (\$100/square foot for a 60-foot ceiling).
- 4) Clean room and cleaning procedures, and training, are necessary to ensure the required spacecraft level cleanliness.
- 5) Existing cleaning and clean room procedures can be adapted to Voyager.

6.0 PHOTOIMAGING CONSIDERATIONS

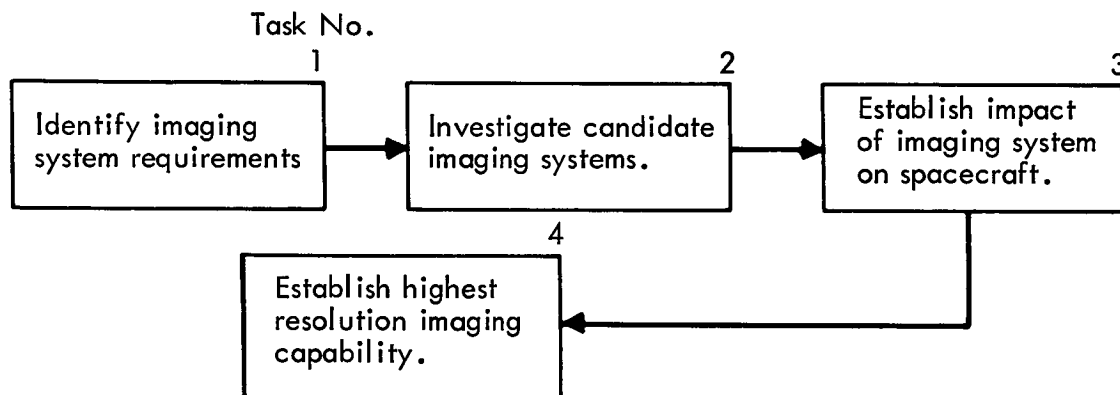
6.1 OBJECTIVES

The objectives of the photoimaging considerations task were to:

- 1) Investigate and compare electrostatic tape, film, and vidicon photoimaging systems.
- 2) Determine the impact of the photoimaging system on the flight spacecraft.
- 3) Establish the highest achievable resolution for each of the photoimaging systems as a function of weight.

6.2 APPROACH

The approach for achieving the three objectives of this study is illustrated below:



The study was accomplished in four tasks as described below:

Task No. 1 -- Photoimaging requirements for each of the 1973 through 1979 Voyager Mars missions were defined in terms of resolution and coverage.

Task No. 2 -- The capability of the three candidate photoimaging systems to satisfy the previously developed photoimaging requirements were examined. The characteristics of these three photoimaging systems then were developed.

Task No. 3 -- The impact of the three photoimaging systems on the flight spacecraft was established.

Task No. 4 -- The resolution capabilities of the three candidate photoimaging systems were compared considering the impact on total spacecraft weight. The highest achievable imaging resolution then was established.

6.3 RESULTS

Study results summarized in this section cover the following:

- 1) Imaging system requirements in terms of both resolution and coverage.
- 2) Conceptual description and gross physical characteristics of the three candidate imaging systems.
- 3) Key optical and imaging sensor parameters in nomographic form to facilitate imaging system analysis.
- 4) Analysis of the three candidate imaging systems -- vidicons, film cameras, and electrostatic tape cameras -- including considerations of resolution, format, stereo, coverage, focal length, aperture, exposure time, optics design, weight, size, smear and image motion compensation, and color imagery.
- 5) Comparison of the three candidate imaging systems on the basis of (1) contribution to mission success and (2) performance of mission objectives including considerations of impact of resolution requirements on imaging equipment spacecraft subsystem weight.

6.3.1 Imaging System Requirements

Photoimaging requirements derive from the basic scientific objectives of planetary exploration. These scientific objectives include (1) search for extraterrestrial life and (2) understanding the evolution of the solar system. Spacecraft-borne photoimaging experiments can contribute to these objectives during Mars orbital operations:

- Locating evidence of present and past favorable environment for life.
- Studying planet surface composition.
- Determining major constructive, destructive, and transport sources.
- Studying isostatic adjustments.
- Studying cloud patterns and storm dynamics.
- Providing map base for geologically-oriented data from other science.
- Surveying potential capsule sites.

Photoimaging coverage and resolution requirements to satisfy these objectives have been developed and are summarized below:

1) Coverage

Extended coverage - 1973: 60°S to 40°N (75% of planet)
(Medium resolution)

1975-1979: Entire planet

Selected coverage - 1973: 0.1% of planet surface
(High resolution)

1975-1979: 1% of planet

Duration - Adequate to examine seasonal changes

1973: 6 months

1975-1979: 2 years

2) Resolution

Medium resolution - 100 to 300 meter resolution for planet surface mapping
(1973-1979)

High resolution - 1 to 10 meter resolution for life detection from orbit,
capsule site survey, and surface characteristics (1973-
1979)

The Mars area coverage requirements (both medium and high resolution) are similar to those obtained by Lunar Orbiter for the moon. The medium resolution requirements also are similar to those obtained by Lunar Orbiter on the high altitude total surface mapping missions (LO IV and LO V). Similar Voyager requirements will allow the construction of Mars maps of sufficient quality to permit geological (areological) studies. The high resolution requirements are keyed to the objective of extraterrestrial life detection. As indicated in Figure 6-1, ground resolution of less than 0.1 to 0.5 meter could probably not be achieved because of atmospheric scattering, turbulence, and haze. However, resolutions on the order of 1 meter will enable the detection of Mars surface objects the size of large Earth mammals (e.g., large whale).

6.3.2 Description of Candidate Imaging Systems

Three photoimaging systems were considered. These were: (1) a silver halide film camera system, (2) a vidicon system, and (3) a dielectric or electrostatic tape system.

The advantages and disadvantages of these three systems are summarized in Figure 6-2. Each of these systems is discussed in the following paragraphs.

6.3.2.1 Silver Halide Film Camera Systems

Silver halide film systems can store a large quantity of data at a high packing density. Film systems were used for Earth orbital and Lunar Orbiter photographic missions. The extension of the film cameras' capability to a Mars planetary mission were considered. The specific systems investigated were: (1) Eastman-Kodak Lunar Orbiter photo unit, (2) a Fairchild Camera planetary film system and (3) the ITEK Voyager photo unit. The Lunar Orbiter photo unit is quite representative and is described below.

The Lunar Orbiter photo subsystem is a dual camera having (1) a 24-inch focal length lens for high resolution imagery, and (2) an 80-mm lens for wide angle, medium resolution mapping imagery. This approach was selected for Lunar Orbiter

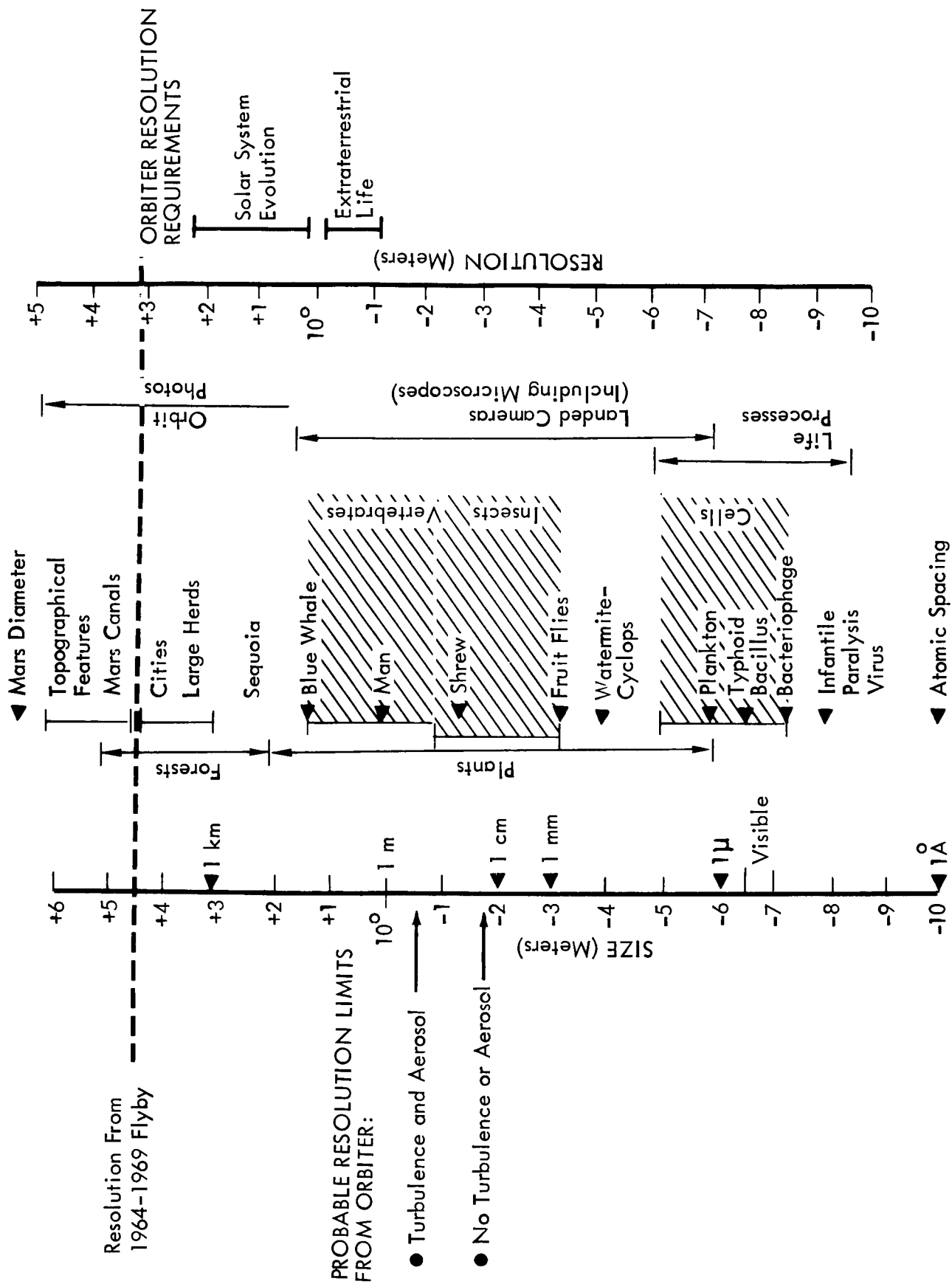


Figure 6-1: IMAGING EXPERIMENT RESOLUTION REQUIREMENTS

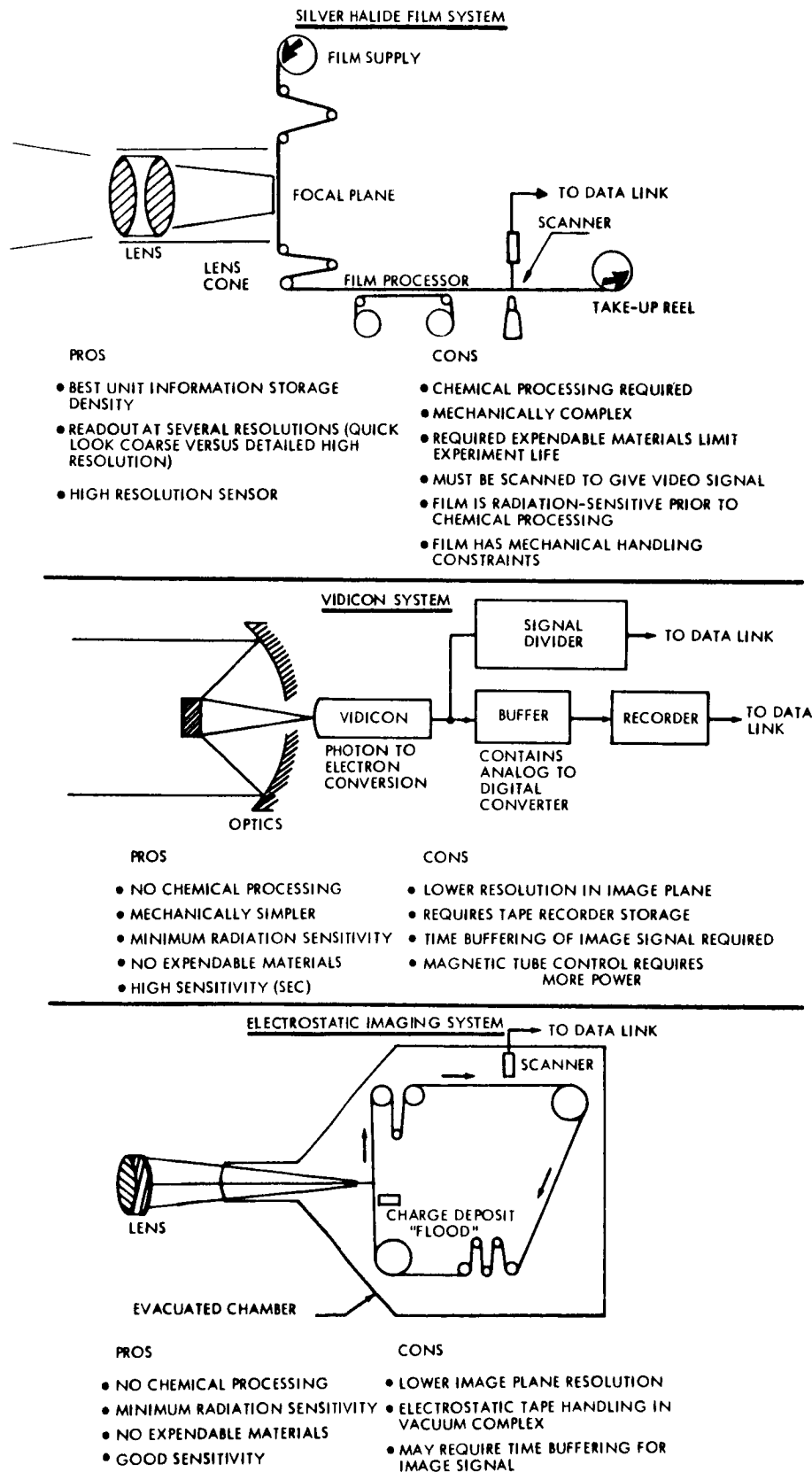


Figure 6-2: CANDIDATE PHOTOIMAGING SYSTEMS FOR THE VOYAGER SPACECRAFT

to satisfy the special requirements of lunar landing site selection and geological analysis. This photo subsystem uses a self-contained monobath web film processor. The scanner uses a specially designed cathode ray tube for a scanning light source. The light transmittal through the film is read with a photomultiplier tube.

The duration of the Voyager mission poses problems in the use of the specific Lunar Orbiter photo subsystem processing unit. The present Bimat film processing technique has a maximum lifetime of about 16 weeks. The Voyager mission lifetime, in excess of a year, requires improvements in the photo processing technique. These improvements are within the current state of the art.

All film systems are radiation-sensitive. Therefore, Voyager will require a small amount of additional shielding to protect the film from the hazards of ionizing radiation.

6.3.2.2 Vidicon Systems

Television systems employing vidicon tubes and tape recorders have been used successfully on Earth orbital, lunar, and planetary space missions. Such systems are highly developed and have demonstrated reliable performance under space environmental conditions. The Mariner IV television system is of special interest since it was designed specifically for a Mars flyby mission. This system includes a 12-inch focal length f/8 Cassegrainian optical system and masked photocathode area of 0.22 x 0.22 inch. This system provided a resolution of about 3 km at the closest Mars approach of 11,500 km. The system has a frame time of 48 seconds and records the images with a magnetic tape recorder having a capacity of 5.2×10^6 bits. This capacity is sufficient to record 21 pictures for later transmission.

An improved version of the vidicon known as the secondary emission conduction (SEC) vidicon is currently being investigated for space applications. This tube contains a planar electron multiplier which takes the accelerated photo electrons from the photocathode, amplifies this current flow by a factor of 100, and produces a new charge pattern. This pattern is read out in a manner similar to the ordinary vidicon. This tube is a vidicon with greatly improved actinic sensitivity. To date the SEC vidicon has not been used in space. One major deterrent to its use is the fragile structure of the secondary electron conduction target. Ruggedizing the SEC vidicon will permit its use for Voyager.

6.3.2.3 Electrostatic Tape Cameras (ESTC)

The electrostatic tape camera, also referred to as the dielectric tape camera, combines the features of the vidicon and tape recorders into an integral storage tape system with electronic readout. Several different types of dielectric tape cameras have been under development. The system developed for Nimbus is the only one sufficiently advanced to warrant consideration for a Voyager mission.

The dielectric tape is manufactured on a flexible cronar base. A conducting medium, usually a copper-gold mixture, is deposited across the tape. Photoconducting material, similar to that used in vidicon targets is then deposited, followed by a layer of insulating material. Such tapes are made in limited lengths (approximately 35 meters) with both 35-mm and 70-mm widths.

The dielectric tape camera is basically a mechanical vidicon. Operation of the dielectric tape camera consists of a sequence of steps: First is that of "erasing", or sensor preparation. In this step old information is erased from the tape, and the surface of the insulator is brought to a uniform, known potential. Upon completion of the prepare cycle, the tape is ready for exposure. During exposure the optical image, as well as a simultaneous raster scan of electrons, is required. On completion of the exposure operation, the optical input and electron scan are terminated. The resulting stored charge pattern is ready for readout or storage until some later time. When readout is required, it is performed by scanning the surface of the insulator with a finely focused electron beam. This readout operation can be performed more than once.

The dielectric tape camera has several features that make it conceptually attractive for planetary applications. The system is all electronic and requires no chemical processing. The tape is reusable, eliminating the need for a large tape supply on long-duration missions. Also, the tape has a low susceptibility to radiation damage. The major system disadvantage occurs in the tape transport mechanics.

6.3.3 Imaging Systems Analyses

6.3.3.1 Resolution and Coverage Considerations

Ground resolution and surface coverage per frame are closely dependent parameters. The requirement for a given ground resolution from a specified orbital altitude fixes the focal length for the imaging system considered. This fixed focal length, combined with mechanical size limitations of the optical sensor, also fixes the surface coverage per exposure frame.

In general, (1) vidicons and electrostatic tape cameras are best suited to square formats; (2) high resolution silver halide film cameras can have up to a 4:1 format aspect ratio; and (3) medium resolution, wide angle mapping cameras are best suited to square formats due to lens design constraints.

Extensive ground coverage without compromising resolution may be obtained by rapid cycling of a particular sensor as the spacecraft progresses in orbit. This procedure, however, creates two problems:

- Data rate saturation from vidicons
- Mechanical film handling rates in the silver halide cameras.

The film handling problem can be solved by proper camera design. The data rate saturation problem associated with the vidicon sensor is severe if large amounts of data are required. The solution to this problem will require state-of-the-art advances in data storage equipment and techniques.

Resolution Considerations--In establishing the optical system parameters for satisfying photoimaging requirements, consideration is first given to attaining the required resolution. A nomograph relating image system, focal length, sensor resolution, and orbital altitude to the required ground resolution is given in Figure 6-3. Sensor resolution is specified in lines/mm measured in the sensor's image plane. The highest spatial frequency, in lines/mm, that can be resolved by the sensor is

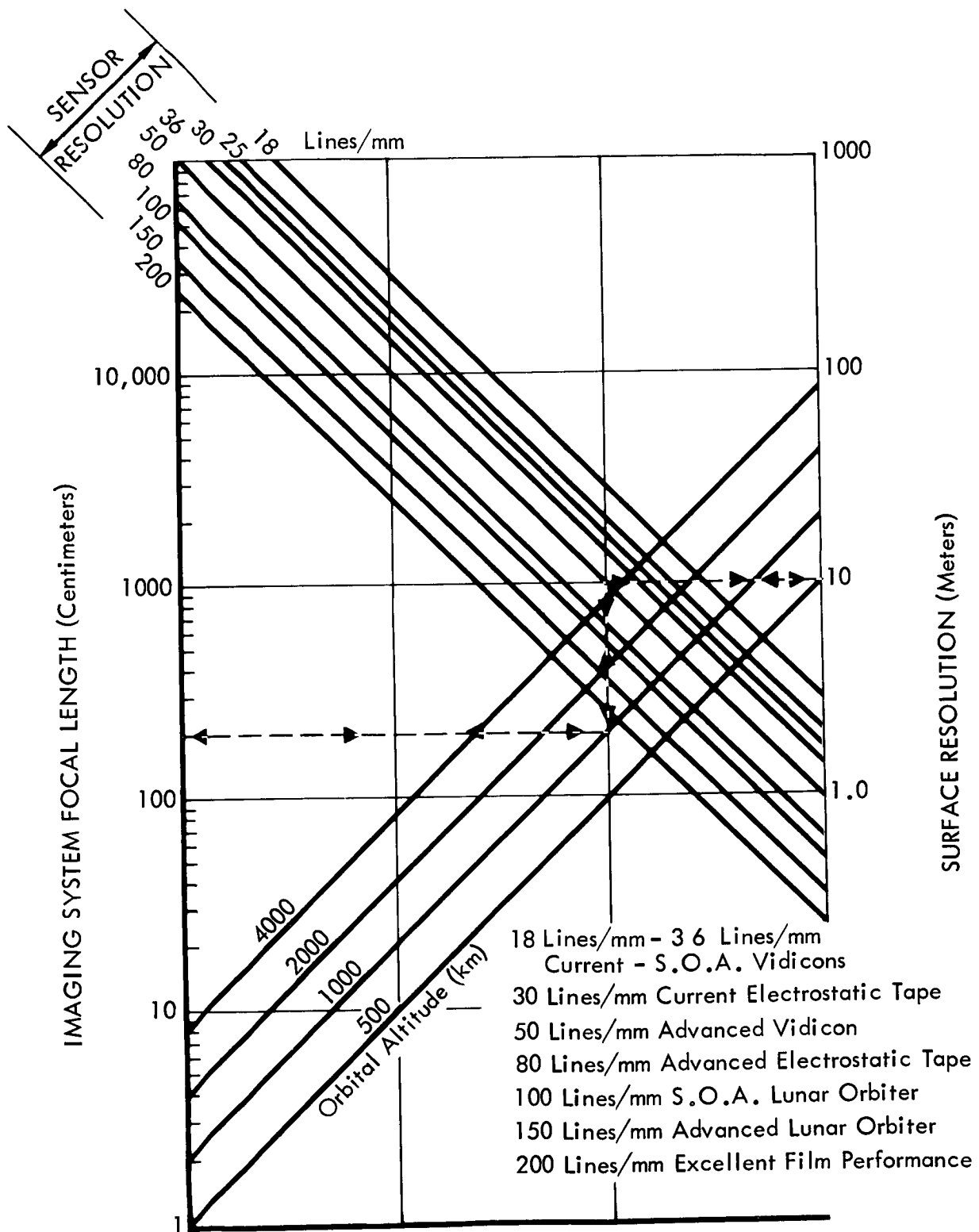


Figure 6-3: SURFACE RESOLUTION NOMOGRAPH

its cutoff frequency. The sensor resolutions cited in the nomograph are the cutoff frequency for a particular system. This means an item imaged at the cited spatial frequency (lines/mm) is at the detection threshold of the system. It is significant to note that the resolution thresholds quoted by a device manufacturer are generally optimistic, especially for a new or forecasted sensor. Therefore, a conservative derating of forecasted sensor performance is recommended.

Based on the medium resolution mapping requirements of 100 to 300 meters from an orbital altitude of 500 km, the following focal lengths are required for the candidate imaging systems:

	Focal Lengths	
	Vidicons and Electrostatic Systems	Silver Halide Systems
100-Meter Resolution	60 mm to 130 mm	30 mm to 60 mm
300-Meter Resolution	20 mm to 45 mm	10 mm to 20 mm

The above focal lengths cover the range from current state of the art to advanced sensors. For these focal lengths, the lenses are generally small, resembling the conventional 35-mm camera lenses.

The exceptions are the lenses for the 10-mm to 20-mm focal lengths. The shortest focal length standard mapping lens with suitable field of view is 76 mm. Experimental versions of this lens have been fabricated in a 38-mm focal length for use with 70-mm format film. Wide angle mapping lenses with shorter focal lengths present serious design and format coverage problems.

In conclusion, the vidicons and electrostatic imaging systems can satisfy the medium resolution requirements with conventional lens design. The inherent resolution of the silver halide film is more than adequate for medium resolution imaging, but may require an unusual design of short focal length mapping lenses.

High resolution imaging for life detection presents a more severe lens focal length problem, as shown in the table:

	Focal Lengths	
	Vidicons and Electrostatic Systems	Silver Halide Systems
1-Meter* Resolution	6 Meters to 13 Meters	3 Meters to 6 Meters
10-Meter* Resolution	60 cm to 130 cm	30 cm to 60 cm

*From 500-km orbital altitude.

The lens design required for achieving a resolution of 10 meters is straightforward. As an example, the Lunar Orbiter camera used a focal length of 60.6 cm for high resolution photography.

The overall length of lens required for achieving a 1-meter resolution can be reduced significantly by using predominantly reflective optics incorporating folded design concepts. Therefore, subsequent analysis will emphasize modified Cassegrainian optics. Such a folded design will still impose severe requirements on the spacecraft due to the size and weight of the lens alone.

Coverage Considerations--Sensor size is the key parameter that determines surface coverage per frame. Sensor sizes for current and advanced candidate imaging systems are illustrated in Figure 6-4.

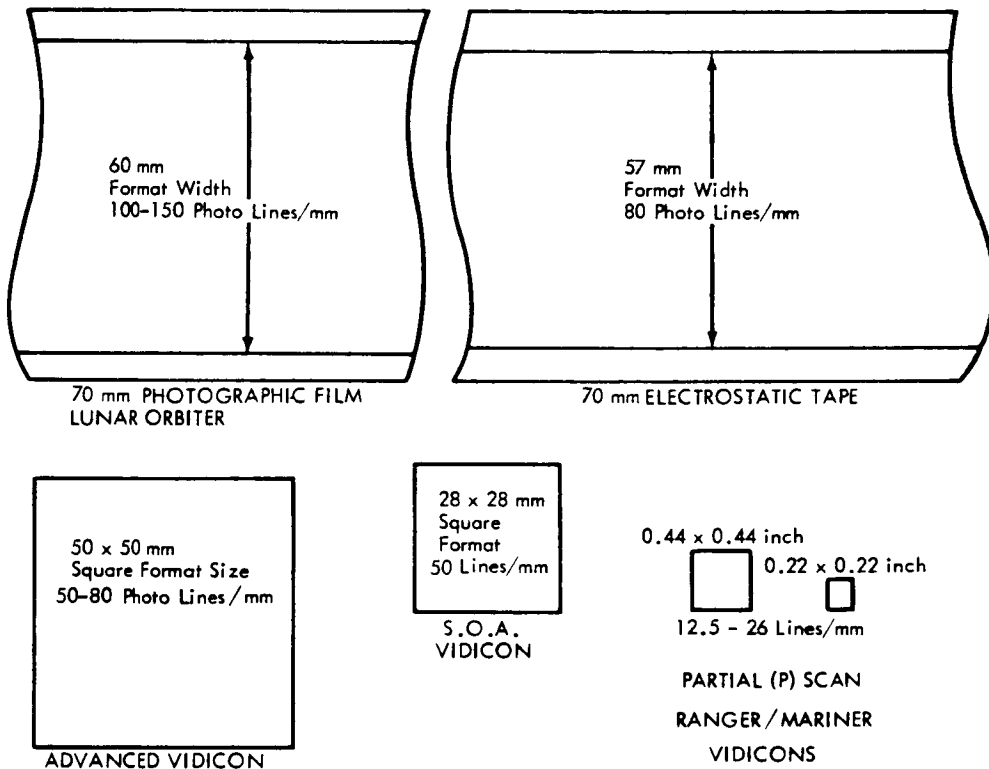


Figure 6-4: SIZE OF CANDIDATE IMAGING SYSTEM SENSORS

A nomograph relating image system focal length, orbital altitude, and sensor dimension to the width of surface coverage is shown in Figure 6-5.

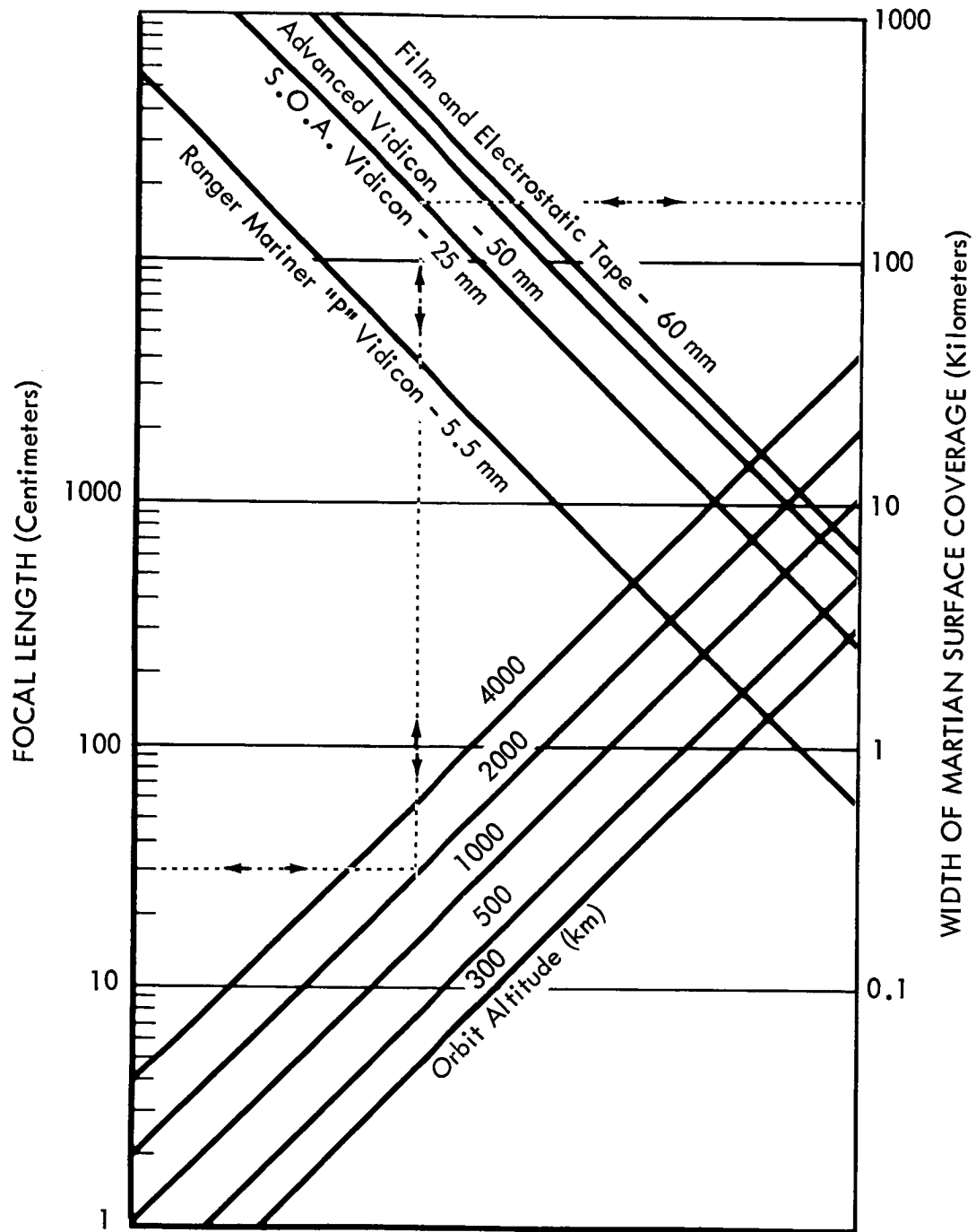


Figure 6-5: SURFACE COVERAGE NOMOGRAPH

Width of coverage for the candidate imaging systems from a 500-km orbital altitude is summarized below:

Resolution	28 x 28 mm Vidicon, 50 lines/mm version	50 x 50 mm Vidicon & electrostatic tape, 80 lines/mm version	70 x 70 mm Silver halide film, 100 lines/mm version
100 meters	130 km x 130 km	400 km x 400 km	600 km x 600 km
300 meters	390 km x 390 km	1200 km x 1200 km	1800 km x 1800 km
1 meter	1.3 km x 1.3 km	4.0 km x 4.0 km	6 km x 24 km (4:1 film format)
10 meters	13 km x 13 km	40 km x 40 km	60 km x 240 km (4:1 film format)

Technological improvements will result in larger formats, thereby increasing ground coverage. The improvement trend in surface coverage width for the candidate imaging systems is shown in Figure 6-6. For a given ground resolution, surface coverage width improvements are obtained by increasing sensor size and sensor resolution. The vidicon is seen to have the most growth capability in both format size and resolution. However, even by 1975, its predicted growth will not match the resolution/coverage capability of current film systems.

Stereo Coverage Considerations--Shadow detail will be present in all the contemplated image data. By knowing the lighting geometry at the moment of exposure, considerable knowledge of height may be derived from a single photo. A stereoscopic system mapping the surface at the required medium resolution of 100 to 300 meters would allow generation of height contours with approximately 160- to 500-meter intervals. Shadow detail analysis of nonstereoscopic medium resolution data should provide comparable height contour accuracy, especially at high phase lighting geometry (90-degree phase is terminator photography).

Redundant coverage every several months is planned for analyzing seasonal changes and life detection. Such coverage can yield limited stereoscopic data for height contouring if the exposures are obtained from sufficiently different orbital camera positions. For the advanced Mars missions, stereoscopic coverage may become a major scientific objective. For satisfying this objective, imaging system redesign is not required. Convergent stereo image data can be obtained for selected sites by either maneuvering the spacecraft or articulating the scan platform to give fore-aft or orbit-to-orbit coverage.

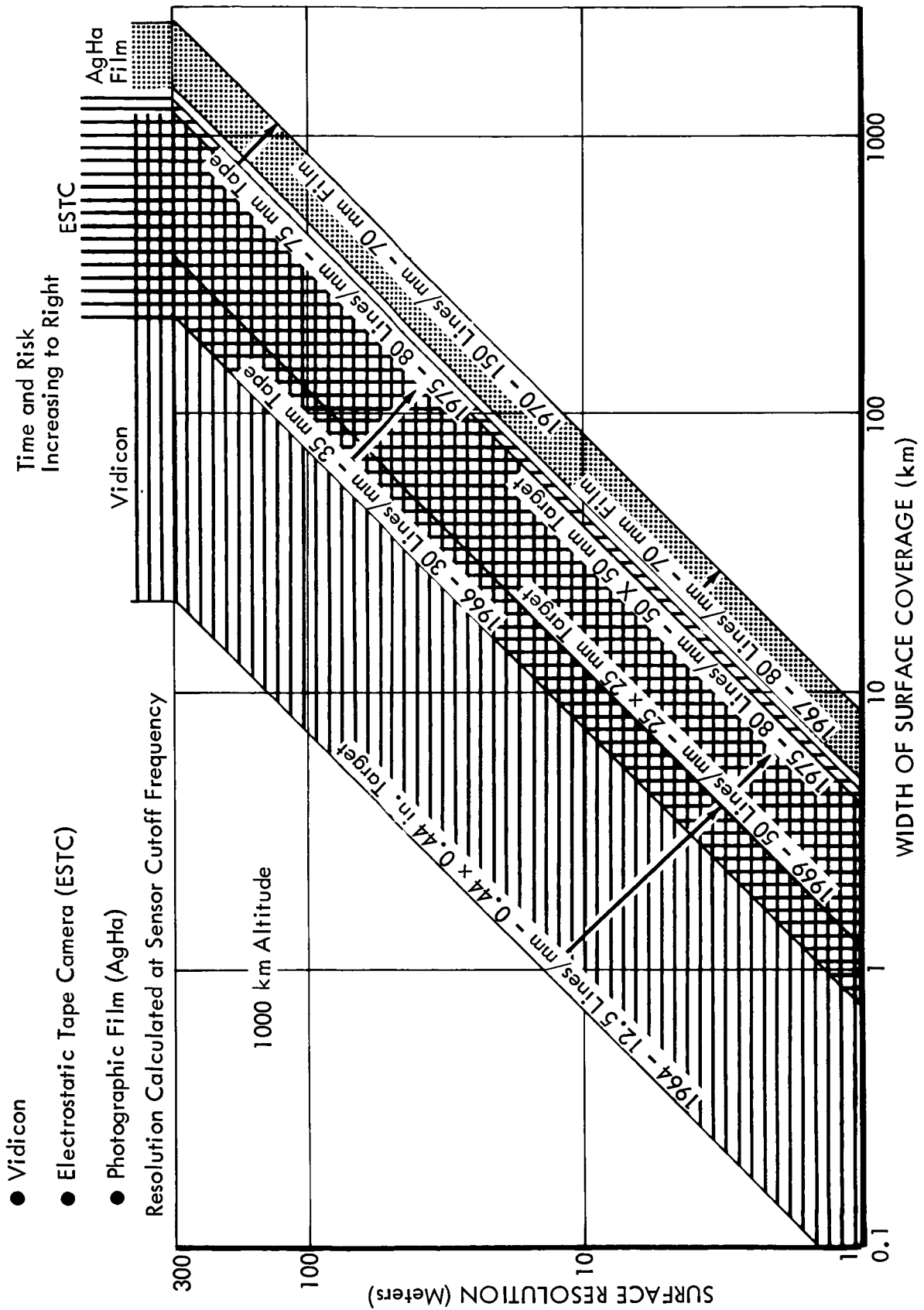


Figure 6-6: WIDTH OF COVERAGE TREND FOR THE THREE CANDIDATE IMAGING SYSTEMS

6.3.3.2 Optical System Parameters

The focal lengths and sensor size to achieve the required ground resolution and surface coverage were established in the previous section. The weight of the optical system is determined not only by the focal length, but by the lens diameter as well. The lens diameter (aperture) is related to sensor sensitivity and exposure time and scene brightness. A nomograph relating lens aperture to these parameters is given in Figure 6-7. Lens aperture is the ratio of lens focal length to diameter, commonly referred to as the f-number. Sensor surface sensitivity is quoted in meter-candle-seconds. Scene brightness is given as a function of the illumination phase angle g . This angle is defined as the angle between the incoming illumination and the image system viewing line. Subsolar imaging occurs at $g = 0$ degrees and terminator imaging occurs at $g = 90$ degrees. For the data shown in Figure 6-7, the image system optical axis is assumed to point along the local vertical. For a given sensor and illumination angle, the required lens aperture is a function of exposure time. For example, a Mariner IV vidicon sensor, with a sensitivity of 0.02 meter-candle seconds will require a lens aperture of $f/4$ with an exposure time of 1.25 milliseconds for photography at a phase angle of 70 degrees.

Lenses in the f-number range from $f/4$ to $f/10$ generally are preferred because their design is straightforward. Lenses with f numbers from $f/1$ to $f/4$ are difficult to design and fabricate at long focal lengths because of size and material problems. Lenses with f-numbers in excess of $f/10$ must be used with caution because their smaller apertures result in diffraction limitation problems. However, lenses in this aperture range are quite suitable to SEC vidicons. These smaller aperture lenses result in significant size/weight saving.

For high resolution imaging, focal lengths as long as 13 meters are required. Such focal lengths with lens apertures in the preferred $f/4$ to $f/10$ range will result in large, heavy optics.

Geologists prefer high phase angles because of the contrast and shadow detail obtained. This desired illumination range averages $g = 50$ to 70 degrees phase with some near terminator imaging. Therefore, the illumination available for imaging will be low, from 6 to 30% of the maximum (subsolar) value.

The Lunar Orbiter SO 243 silver halide films will require large apertures and slow exposure speeds. This imposes severe structural, stability, and weight problems for systems with focal lengths in excess of 1 meter. Use of more sensitive film similar to Eastman Kodak SO 226 may be possible with additional radiation shielding. This class of film has four times the sensitivity of SO 243 with some loss of resolution. Such an increase of sensitivity alleviates the structural stability and weight problem cited above.

The SEC vidicons will have sensitivities allowing use of approximately $f/15$ to $f/20$ optics. This will minimize size and weight for long focal lengths. Concurrently, the SEC vidicon also will allow short exposures, thereby minimizing smear. Thus the SEC vidicon is an excellent candidate for a compact high resolution vidicon system.

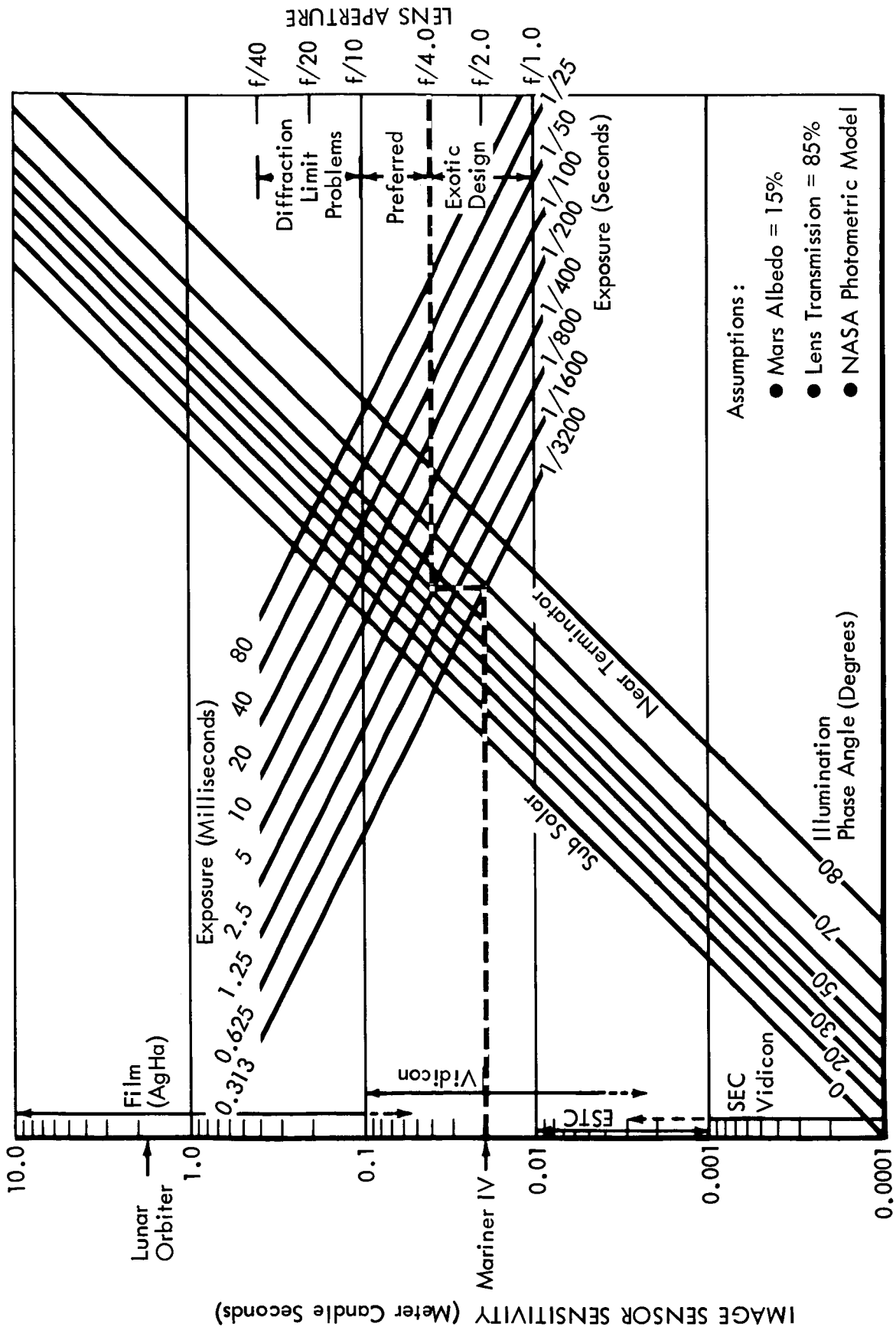


Figure 6-7: LENS APERTURE NOMOGRAPH

Design Considerations for Large Optical Systems -- There are several optical designs that could be considered for large optics. These can be categorized as refractive and reflective designs. Refractive designs become impractical when heavy, large diameter elements of high quality optical glass are required.

The final choice between refractive and reflective designs cannot be made at this time. However, reflective designs are generally lighter, but are more critical to align than refractive designs. Lightweight mirror techniques have made certain reflective optical designs attractive. Weight reductions of about 3:1 over comparable solid mirrors are possible. Even greater weight reductions are feasible using ribbed structures that allow greater diameter-to-thickness ratios for the primary mirror.

Considerable effort has been devoted to the design of reflective optics for space applications.

Refinements of the basic reflective Cassegrainian design appear most suitable for Voyager application. One of their desirable features is a reduction of the overall length of the optics by more than 50% of the focal length. In its basic form, this design consists of a parabolic primary mirror and a hyperbolic secondary mirror, as shown schematically in Figure 6-8.

The field of view of the Cassegrainian telescope is normally limited by coma aberrations. The field of view can be increased by using a nonconic optical surface (i.e., aspherizing the mirror) and adding refractive correcting elements in the converging beam from the secondary mirror. Such a design, illustrated in Figure 6-9, is known as the Ritchey-Chretien telescope. The Ritchey-Chretien telescope designed by Perkin Elmer for the Orbiting Astronomical Observatory is shown in Figure 6-10. The 1-meter diameter f/10 aperture is comparable to that required for the high resolution Voyager imagery.

Weight and Size Estimates for Large Optical Systems -- Weight and size estimates were developed primarily to the Ritchey-Chretien telescope design.

The length of an optical system for high resolution Voyager photoimagery is shown in Figure 6-11, as a function of focal length and f numbers. Primary mirror diameter also is indicated. Optical system length is defined here as the distance from the secondary mirror to the image plane. The volume of the optical system as a function of focal length and f-number is given in Figure 6-12. The total structure volume is approximately 30% larger than that given for the optical system volume.

To estimate optical system weight, the lens was considered to include three basic components. These were: (1) the barrel and baffles, (2) the primary mirror and its mount, and (3) the secondary mirror and mount. Scaling curves for the barrel and baffles were obtained from the OAO model telescope, which used a lightweight construction titanium barrel, and a 1-meter focal length f/5 television telescope developed by JPL for a lunar spacecraft. This television telescope has a 20-cm diameter lens, and a 30-cm long, 2-mm thick, fused quartz barrel. The weight of the primary and secondary mirrors were determined from Figure 6-13 for both solid and lightweight constructions. The weight of the mirror mount was taken as 25% of the weight of the mirror itself. The diameter of the secondary mirror, required

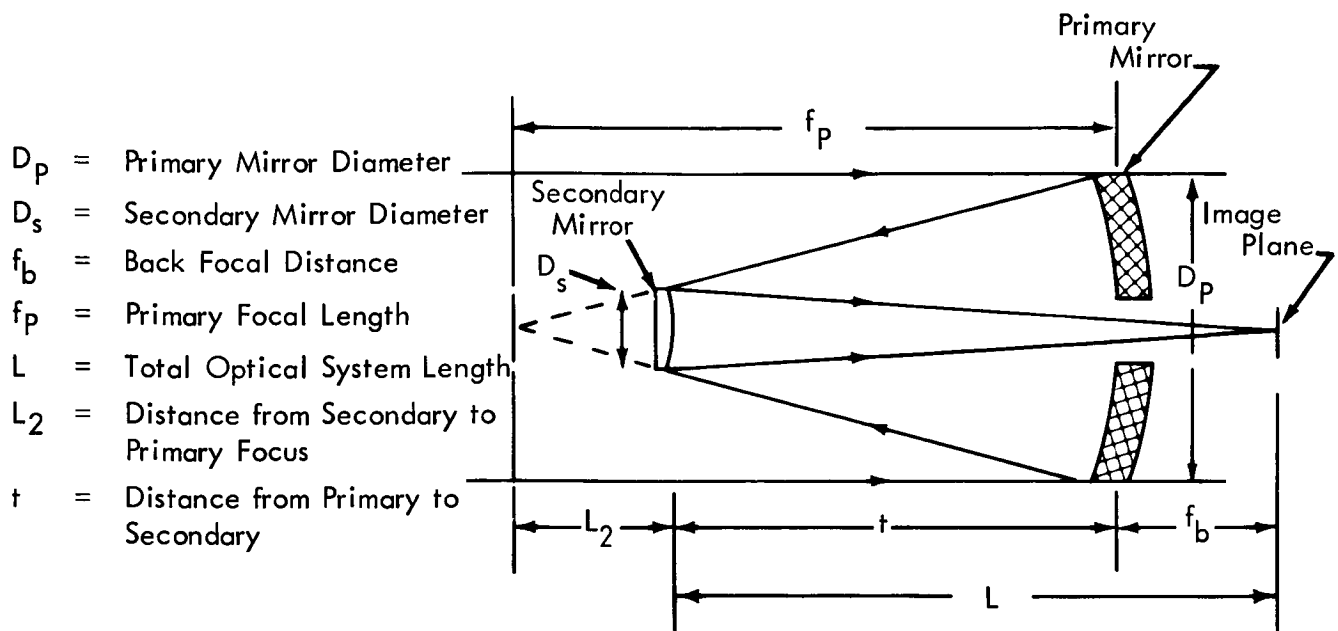


Figure 6-8: OPTICAL SCHEMATIC OF A CASSEGRAIN OPTICAL DESIGN

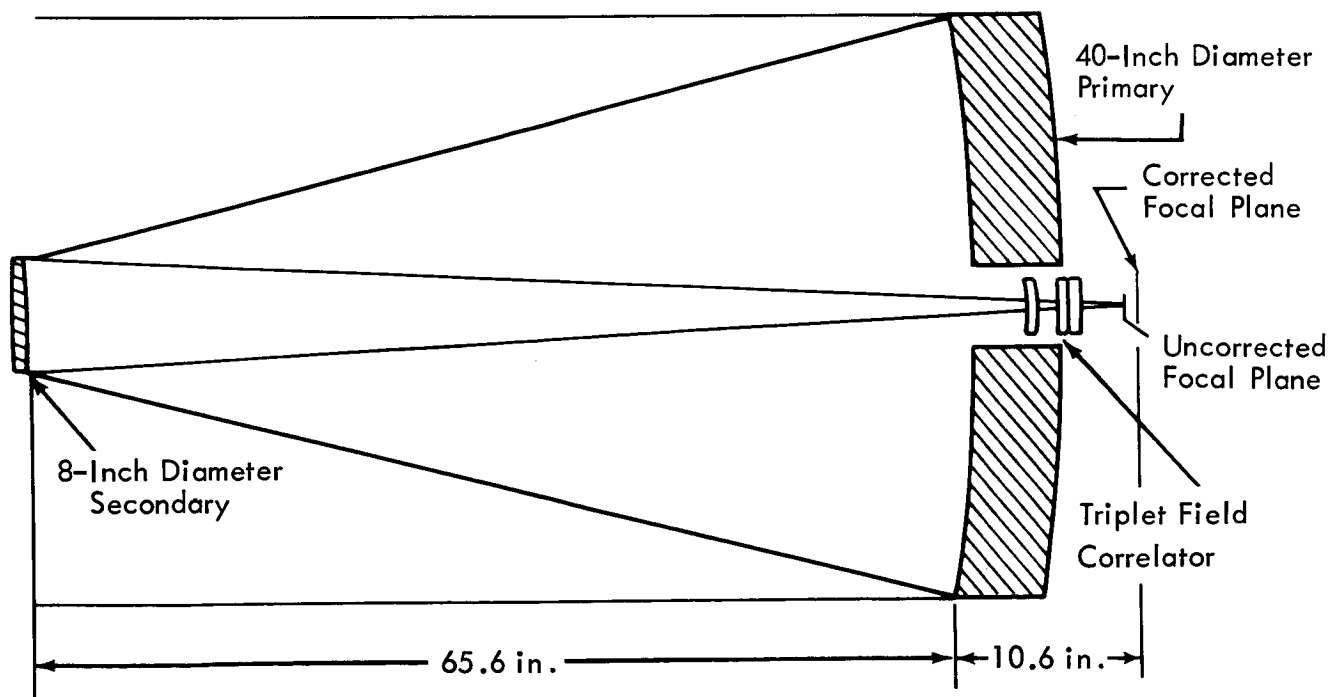


Figure 6-9: RITCHEY - CHRETIEN TELESCOPE AND CORRECTOR DESIGN

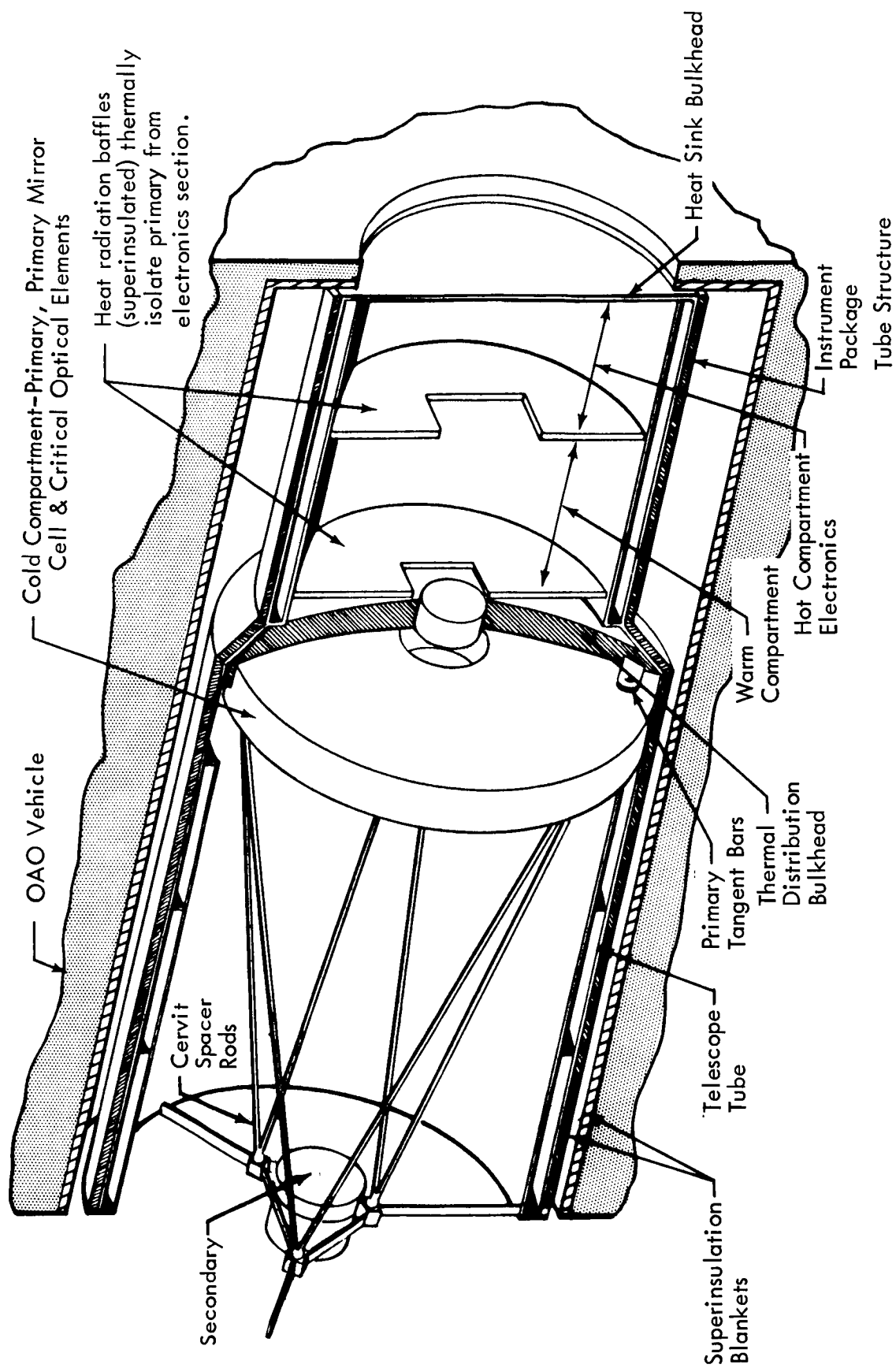


Figure 6-10: RITCHIEY-CHRETIEN TELESCOPE AND CORRECTOR FOR OAO

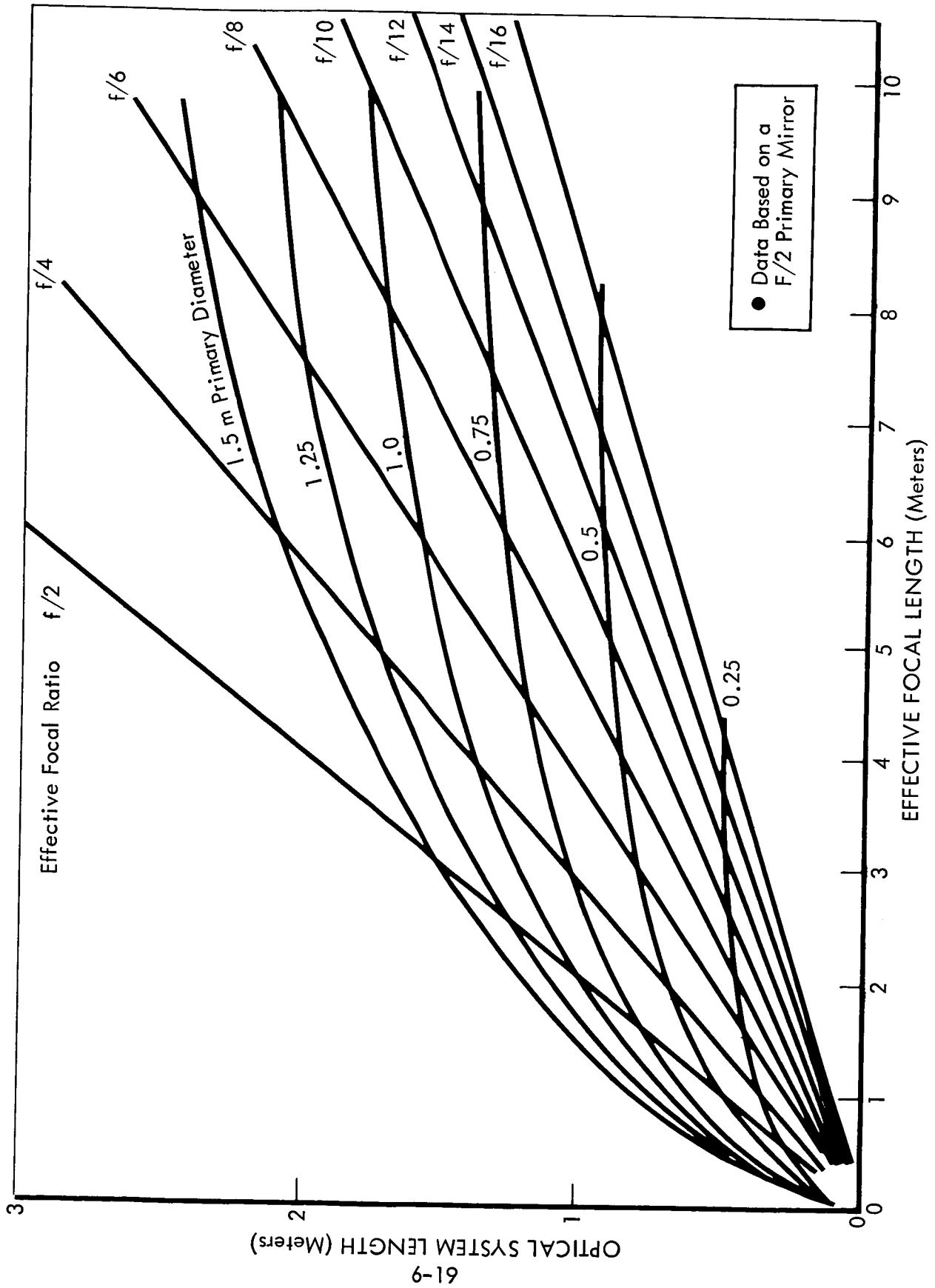


Figure 6-11: TOTAL OPTICAL SYSTEM LENGTH VERSUS FOCAL LENGTH FOR VARIOUS LENS DIAMETERS

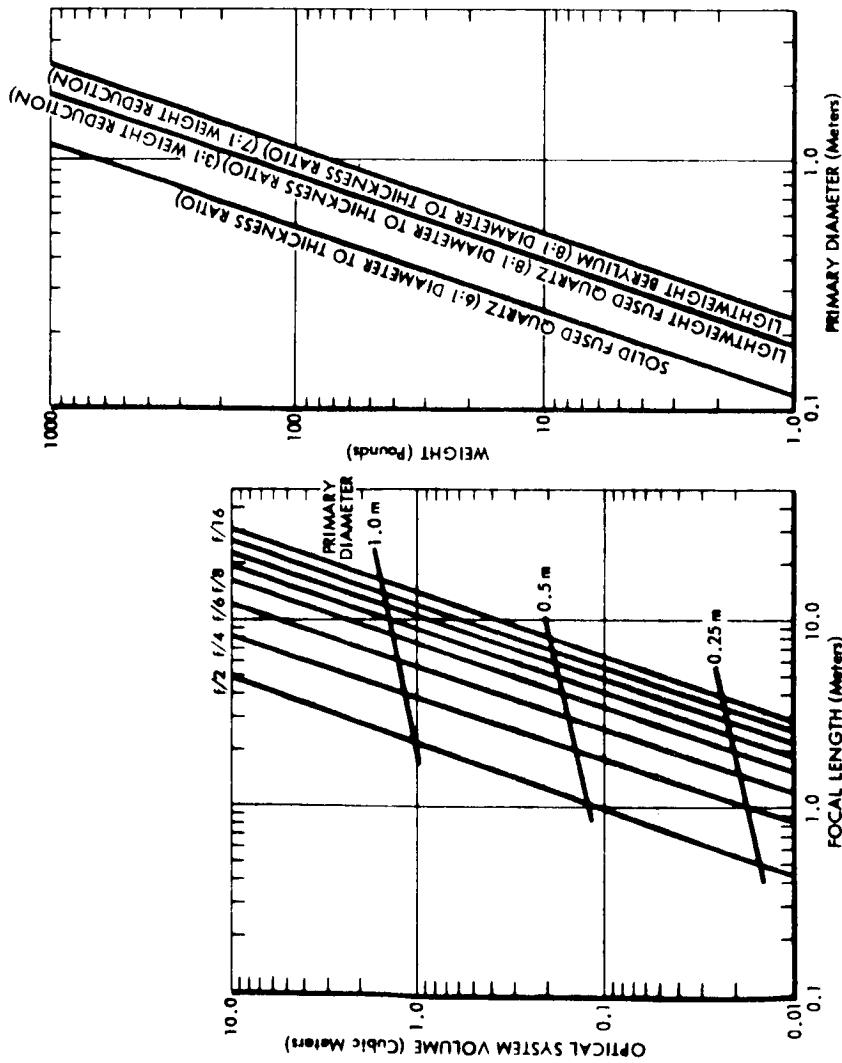


Figure 6-12: FOCAL LENGTH VERSUS VOLUME FOR VARIOUS f-NUMBERS

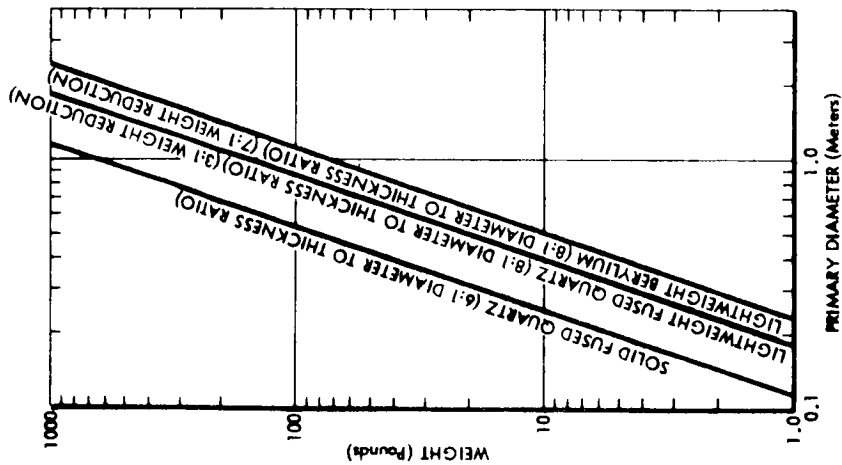


Figure 6-13: MIRROR WEIGHT

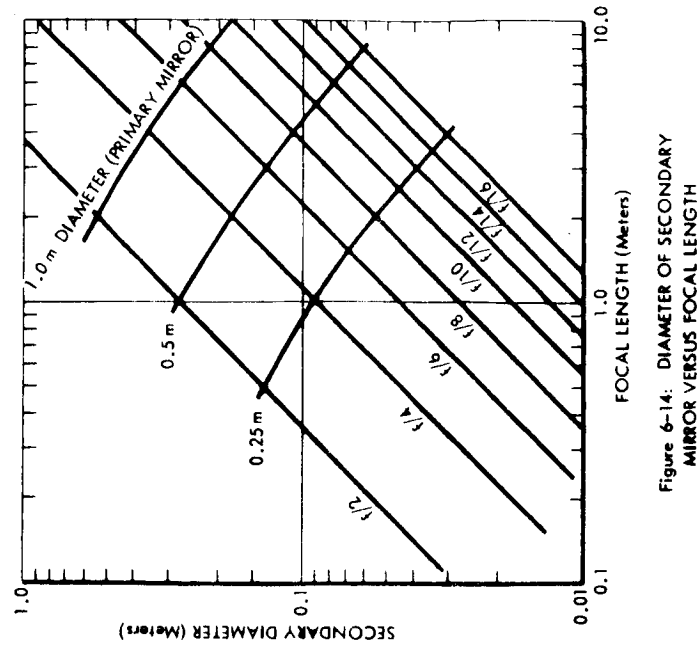


Figure 6-14: DIAMETER OF SECONDARY MIRROR VERSUS FOCAL LENGTH

to establish its weight, is given in Figure 6-14. Total optical system weight is summarized in Figure 6-15, as a function of focal length.

6.3.3.3 Image Smear and Resolution Loss

Motion of an image during exposure will cause resolution loss. Image motion results from the following:

- 1) Spacecraft orbital velocity (approximately 4000 m/sec).
- 2) Mars rotational velocity (approximately 235 m/sec at the equator and decreasing as the cosine of latitude).
- 3) Spacecraft limit cycle rates (0.001 deg/sec when averaged over the limit cycle deadband).

These velocities are illustrated in Figure 6-16 for a 60-degree orbit inclination at its equatorial crossing. It is evident that the orbital velocity is the dominant contributor to image motion.

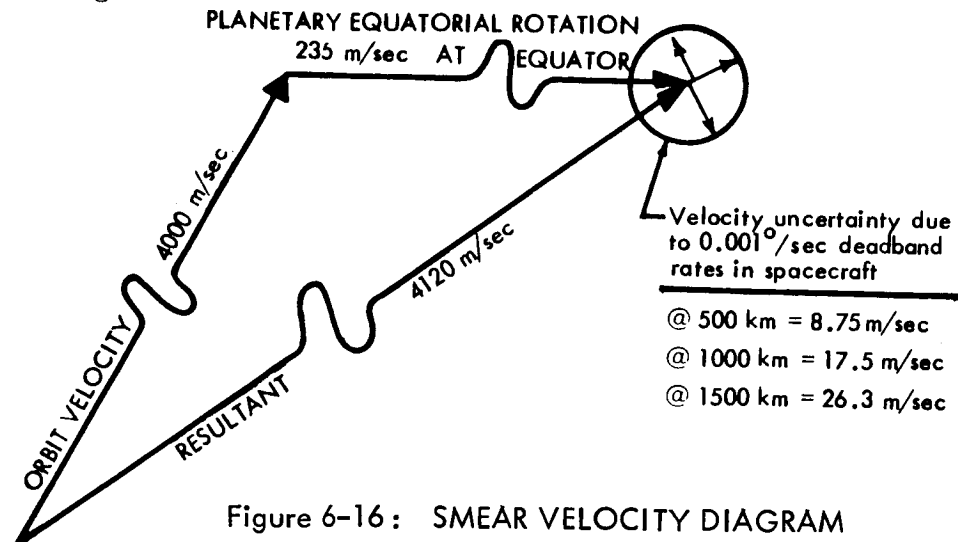


Figure 6-16: SMEAR VELOCITY DIAGRAM

Three practical techniques exist for minimizing the effect of motion-induced image smear or resolution. They are:

- Short exposure times.
- Image motion compensation (IMC) during exposure on the basis of predicted position and velocity.
- Image motion compensation using on-board equipment for measuring the ratio of velocity to altitude (V/H). This was the technique used on Lunar Orbiter.

The high resolution films suitable for Voyager applications are relatively insensitive. Consequently, film cameras may require long exposure times. Hence film cameras for Voyager will probably require image motion compensation for both the high and medium resolution photography. This can be accomplished by moving the film during exposure.

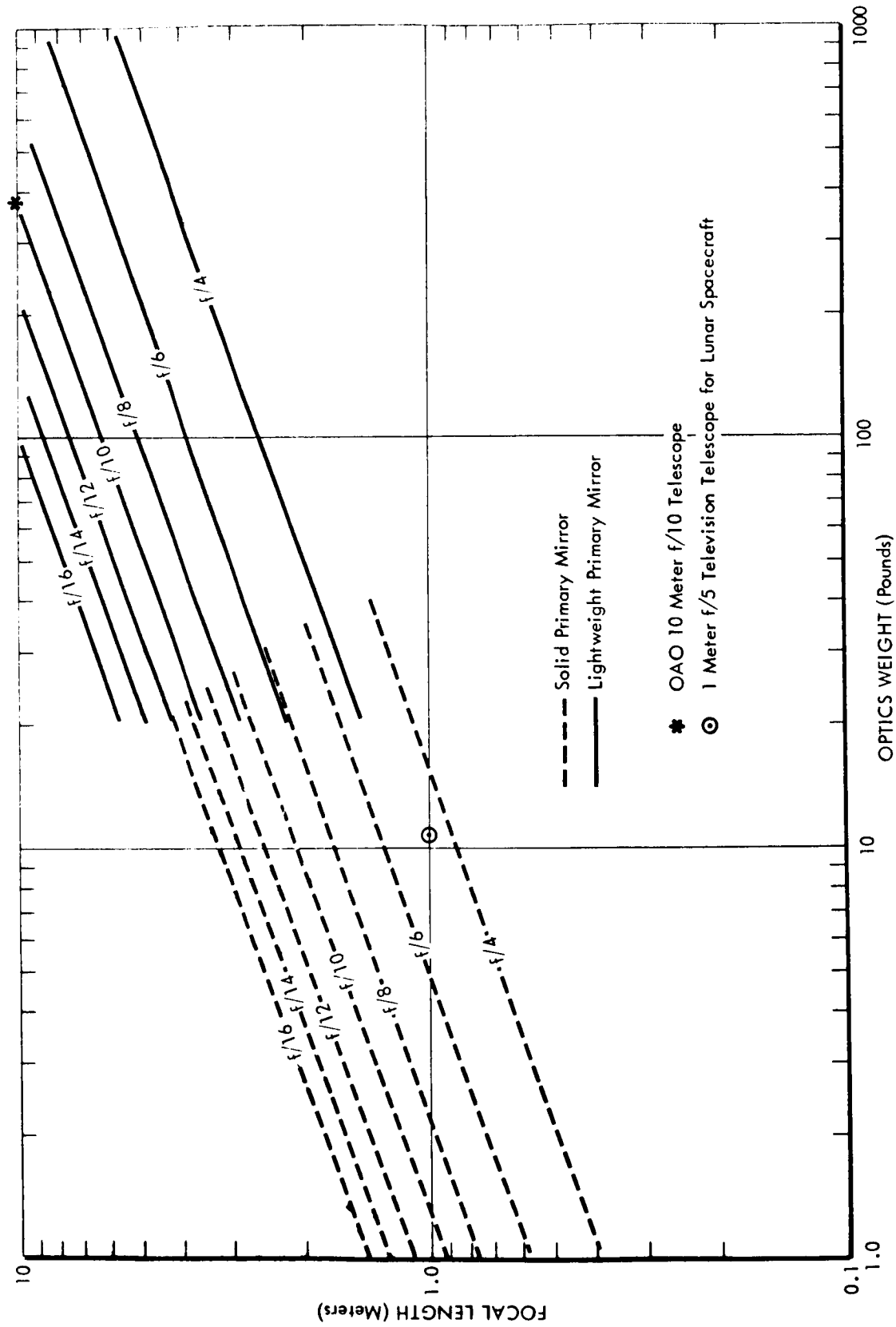


Figure 6-15: OPTICS WEIGHT VERSUS FOCAL LENGTH FOR VARIOUS f-NUMBERS

Voyager vidicon systems for medium resolution imagery can be built with exposure times in the millisecond range. Consequently, image motion compensation will probably not be required. Voyager vidicon systems for high resolution imagery will require IMC. A state-of-the-art improvement in vidicons containing a linear image amplification stage (e.g., a SEC vidicon) may allow electronic IMC with no mechanical motion.

The degradation effects of smear on resolution are shown in Figure 6-17. The curve is empirical and is based on extensive published data. The curve relates the resolution achieved under dynamic conditions to the smear of the target during exposure. As indicated by the curve, smear is negligible when the amount of smear during exposure is less than approximately one-fourth to one-third of the static resolution element. For such smear, the dynamic resolution element is only 10% to 18% greater than the static resolution element.

The existing Lunar Orbiter camera, if used on Voyager would be optically capable of 10 and 80 meters static resolution from 500-km height for the telephoto and wide angle systems, respectively. However, the lighting conditions would require an exposure time of 0.1 to 0.04 second. If IMC were provided by predicting, rather than by measuring, the ratio of velocity to altitude (V/H), a residual smear as large as 5 meters would exist.

This smear is 50% of the telephoto static resolution and 6.25% of the wide angle resolution capability. The smear degradation curve (Figure 6-17) shows the telephoto system (50% case) would be degraded to 129%. Since its static capability is 10 meters, the smear case yields approximately 13 meters resolution for the dynamic case. However, the 80-meter resolution wide angle system is degraded to only 102%, or from 80 to 81.6 meters resolution for the dynamic case.

Alternatively, for the above example, altitude and velocity could have been measured as was done on Lunar Orbiter rather than predicted. Systems capable of measuring V/H and implementing IMC to an accuracy of 99.9% are considered current state of the art. They generally operate only in one preferred direction, that of the dominant (orbital) motion. An error of 0.1% in the system allows a velocity error residual of 4.12 m/sec due to orbit velocity and planet rotation. The velocity due to spacecraft attitude deadband rates would be unmeasurable in general by the V/H detector due to their random orientation and the V/H sensor time constants.

This spacecraft deadband rate contributes a random velocity error of 8.75 m/sec from 500-km orbital altitude. These random residuals can be minimized only by shortening exposure time.

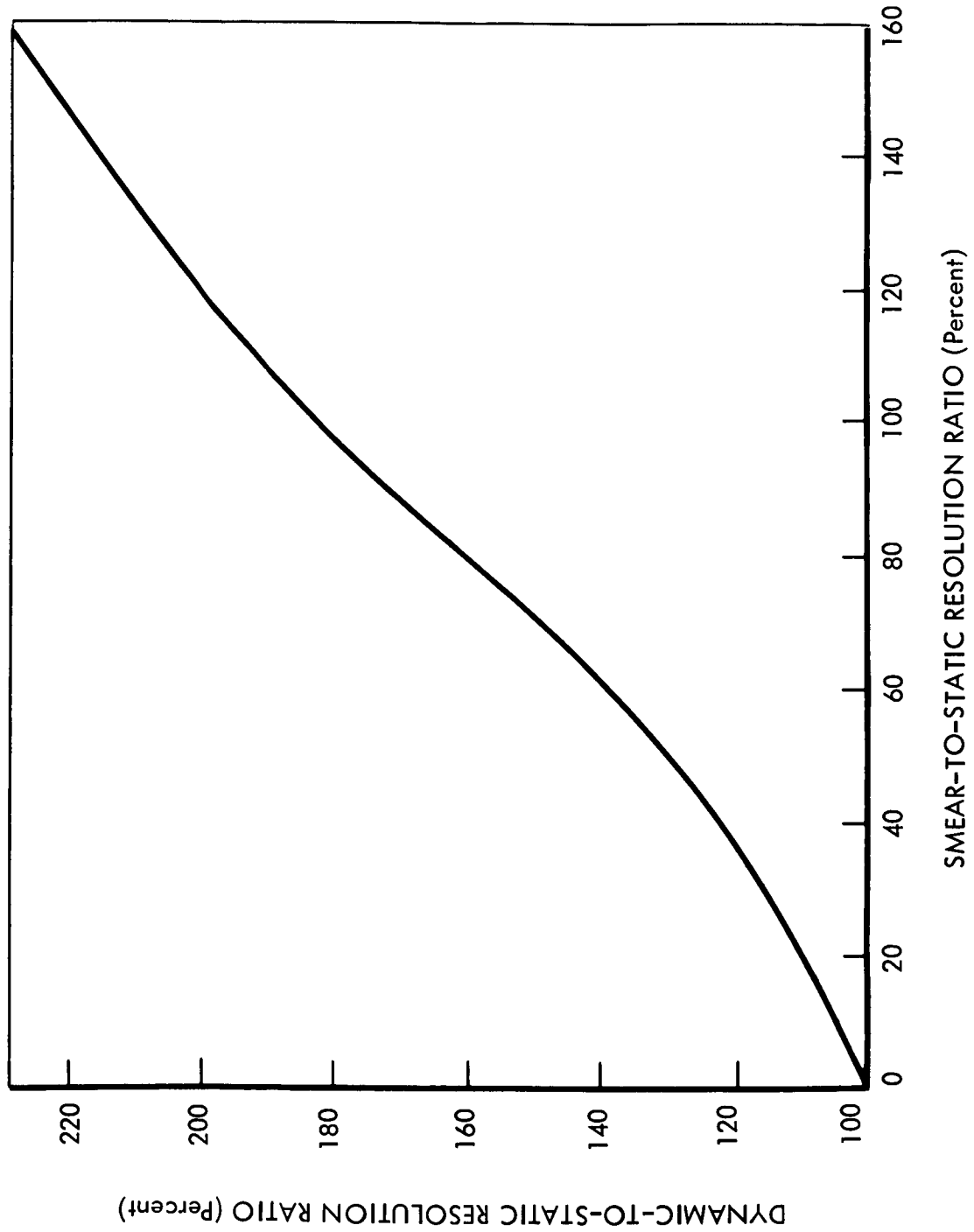


Figure 6-17: SMEAR EFFECTS ON RESOLUTION

6.3.3.4 Color Imaging

Limited color imaging of select Mars surface features, although not specified for the 1973 mission, appears desirable. For color imaging, smear can be significant because of longer exposure times as discussed below.

Two forms of color imaging may be considered for Mars; they are:

Conventional Color -- This form of color would have the same utility to the geologist and biologist as color photographs of terrestrial geological formations and large biological formations (e.g., moss, lichens, algae). To facilitate photo interpretation, maximum color fidelity should be a goal. Because of the Martian atmosphere, some blue scattering is present. Therefore, limited "minus-blue" filtration will be required. This is not detrimental because terrestrial photo geologists, forestry photo interpreters, and biologists are familiar with such "minus-blue" filtration of color aerial reconnaissance photography.

Maximum color fidelity may allow the photo interpreter to recognize features similar to those he has observed terrestrially.

Scientific Color -- This form of color imaging is found in camouflage detection color films. Camouflage detection color films are very sensitive to subtle differences in plant life, foliage, and moisture content. The scientific returns from this form of imaging may exceed those from conventional color, particularly in the area of life detection from orbit.

Both conventional and scientific color may be achieved using four filters in the range from 4200 Å to 8200 Å. This spectral band is within the scope of normal optics and extended range films and vidicons.

Color Imaging Problems -- Key color imaging problems are (1) reduced area coverage and (2) longer exposure time.

- 1) Area Coverage -- Color imaging requires multiple exposures of the same area taken through various color spectral filters. Classical methods, such as mechanically rotating filter wheels, are relatively simple and have been used to great success with vidicons (Mariner IV and Surveyor). Aerial reconnaissance systems use extended spectral range black and white film and mechanical filter wheels or multiple lens arrays.

For Mars, the major problem is the bandwidth available for image transmission. Because of the coverage redundancy required for color, coverage of a large area in black and white is traded against color coverage of an area one-third to one-fourth the size of the black and white area. Hence, color capability may be included as an optional camera feature to be used only for areas of special interest.

- 2) Longer Exposure Times -- Color filtration involves rejection of selected portions of the available imaging spectrum. Therefore, the imaging system

requires either large optics for additional light gathering or longer exposure time for adequate imaging. Spacecraft constraints will limit both weight and volume available for the optics. Therefore, use of longer exposure time for color is the probable solution. It is significant therefore to consider the primary difference between additive and subtractive color imaging, since their exposure times are different.

Additive color (the classical J. C. Maxwell experiment of 1861) projects red, green, and blue light, or images, to form the remaining colors. During the picture-taking phase of an additive system, two-thirds of the available light is removed by filtration. Thus, one-third of the available light is used for imaging. Exposure speeds for this method are at least three times as long as the equivalent black and white image exposure. The long exposures will result in undesirable smear degradation to resolution, especially in the higher resolution systems.

Subtractive color (most familiar as Kodacolor film) employs filters that remove only one-third of the useful spectrum during exposure. These filters are:

<u>Color Absorbed</u>	<u>Filter Name</u>	<u>Appearance</u>
Green	Magenta	Bluish-red
Blue	Yellow	Yellow
Red	Cyan	Blue-green

Since two-thirds of the available light is used for each color imaging exposure, the increased exposure time is only 1.5 over the equivalent black and white imaging. Thus, the smear-during-exposure problem is only 1.5 times that in the noncolor case and only half as severe as with additive color.

The following conclusions are derived from the color imaging considerations:

- Color imaging is obtained at the expense of area coverage due to the required redundancy of imaging.
- Subtractive color processes are more efficient in light gathering, thus minimizing the size/weight impact of a color imaging experiment.
- Both conventional and scientific (extended infrared) color imaging should be considered.
- Optional use of color filtering of selected areas of interest is recommended for current transmission bandwidth constraints.
- A four-color filtration system can satisfy both the conventional and scientific color imaging requirements.

6.3.4 Comparison of Candidate Imaging System

Criteria were established for comparing the three candidate imaging systems. These criteria in order of importance, are as follows:

- 1) Contribution to mission success
- 2) Performance of mission objectives.

Cost considerations were not included in this study.

6.3.4.1 Contribution to Mission Success

Mission success is affected by (1) hardware reliability, (2) compatibility with mission environment, (3) hardware availability, and (4) impact of the imaging system on the reliability of the flight spacecraft.

- 1) Reliability -- The vidicon system appears more reliable than either the film or tape systems. It is simple in design and has no moving parts. Film systems are complex and contain moving parts. Even so, they have operated reliably on space missions such as Lunar Orbiter.

The long duration of the Voyager mission poses problems for the chemically processed film. The Bimat film process (Eastman-Kodak) used for the Lunar Orbiter is limited to approximately a 16-week space mission. Anticipated Bimat improvements will enable missions of up to 1-year's duration. An alternate web film process, Poromat (Fairchild-Camera Corporation) has been successfully stored for 26 months under simulated space conditions. Viscous monobath processing (ITEK Corp.) has an estimated storage lifetime of 36 months. On this basis it is concluded that a film system can satisfy Voyager long life requirements.

The electrostatic tape system is not fully developed. Its operation in vacuum is degraded by outgassing of organic material from the tape and precision bearing long life problems.

- 2) Compatibility with Mission Environment -- Radiation is the key environmental parameter influencing Voyager imaging equipment. All three candidate image sensors are affected by radiation arising from cosmic and solar event phenomena. The general effect of radiation is loss of resolution and image data. Specifically,
 - Vidicons will lose image quality if intense radiation is present during (1) target exposure and (2) slow scan readout. Permanent tube damage is not a problem for the total radiation levels expected.
 - Electrostatic tape systems losses are the same as those for vidicon during image exposure. Gradual destruction of the electrostatic image stored on the dielectric tape also will occur.

- o Unprocessed silver halide films will fog as a result of cumulative radiation until they become totally unusable.

Degradation from radiation may be minimized by shielding the sensor. In the case of a vidicon, the loss of a few frames of data may not warrant the penalty of shielding weight. However, shielding is mandatory for silver halide film.

Data from Eastman Kodak suggests protecting SO-243 film to a maximum integrated radiation of approximately 80 rads. Figure 6-18 shows that 8 gms/cm² of shielding is adequate for a 90-day orbital mission plus Earth-Mars transit. A conservative design goal for SO-243 protection of 30 to 40 rads requires 16 gms/cm² to 20 gms/cm². Nine to twelve pounds of shielding should be adequate protection for a small 100-foot cassette of 70-mm silver halide film. Considerable additional weight penalty is incurred if the camera design requires protection of a film looper storage area. This zone, normally a rather large volume, holds film between the exposure portion of the camera and the film processor.

The data shown in Figure 6-19 indicate that dielectric tape loss of information will not occur for the worst-case total radiation expected for the Voyager mission.

On the basis of the foregoing it is concluded that: (1) vidicons and tape cameras are compatible with their mission environment, and (2) film systems can be made compatible with the mission environment with a small amount of added radiation shielding.

- 3) Imaging Equipment Availability -- Optics and sensors for the medium resolution experiment can be obtained by modifying existing hardware and are considered available. Optics for the high resolution experiment will be similar in design to the OAO Ritchey-Chretien telescope and should be available in time for the Voyager missions. Improved high resolution sensors for the three candidate systems are in development.

Silver halide films probably will remain with the 70-mm format width for space applications. Mechanical handling problems in the areas of processing and image scanning dictate this width constraint. Major improvements anticipated are:

- Improved overall system resolution, increasing from the current 80 lines/mm for the Lunar Orbiter concept to 150 lines/mm about 1970.
- Somewhat improved film sensitivity without a significant resolution loss.
- Simplification of the processing chemistry problems, predominantly in the areas of materials handling.

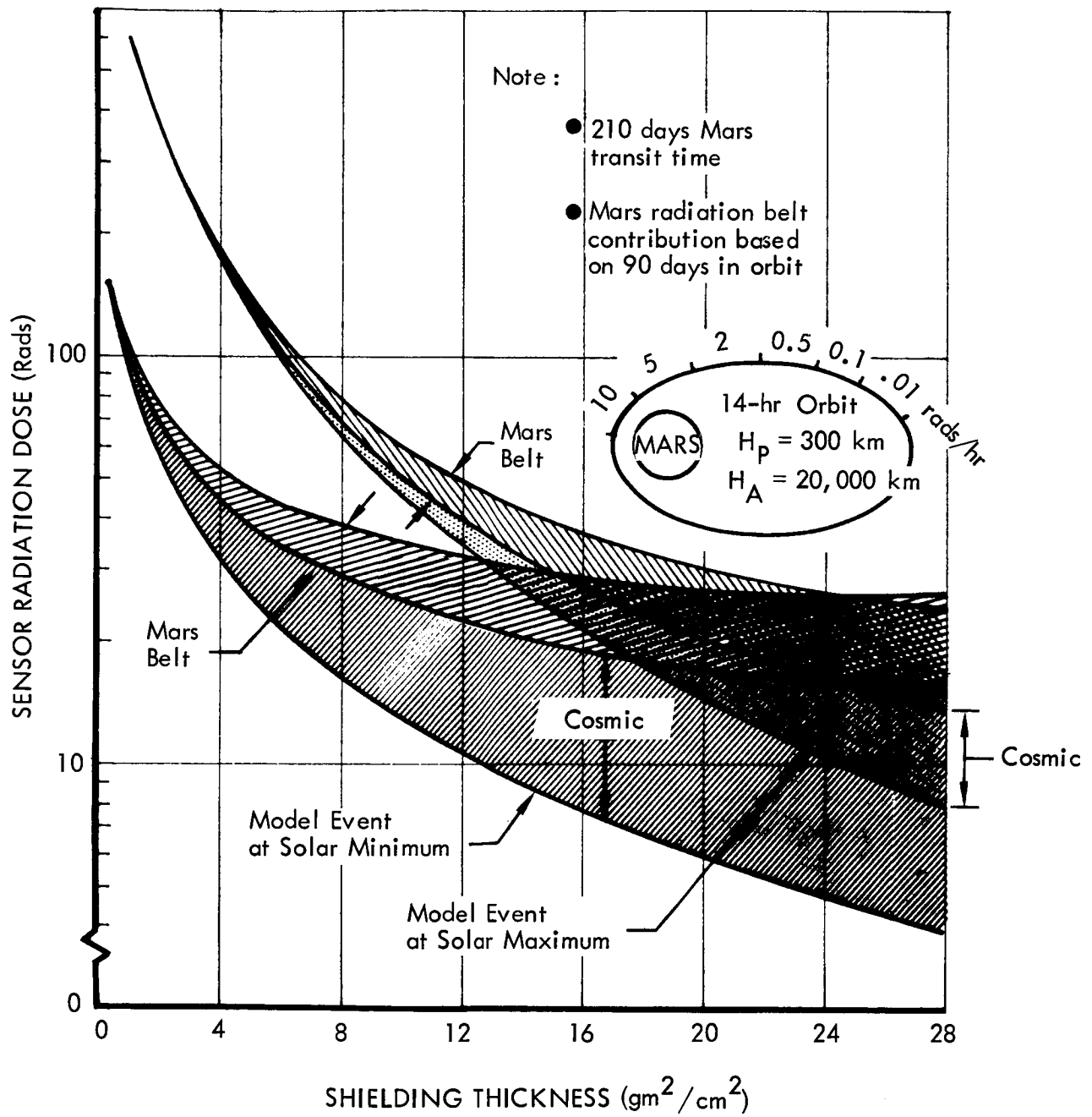


Figure 6-18: SENSOR RADIATION DOSE FOR MARS MISSIONS

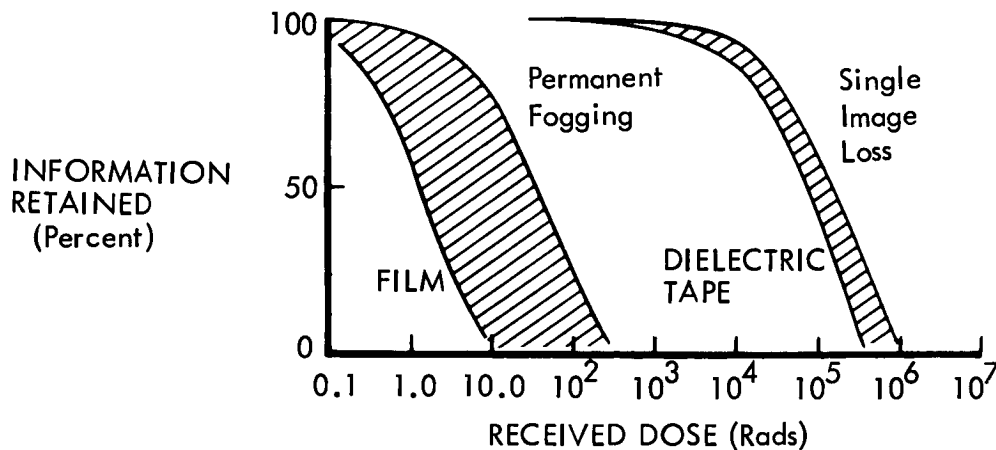


Figure 6-19: COMPARISON OF IMAGE DEGRADATION IN FILMS AND DIELECTRIC TAPE UNDER SPACE RADIATION (RCA DATA)

The electrostatic tape camera concept reached a development plateau with the Nimbus camera. This device uses a 35-mm format width and yields 30 lines/mm at good sensitivity, comparable to the better vidicons. Forecasts suggest a 70-mm version yielding 80 lines/mm would be possible by 1975 if funding were made available.

Larger target area vidicons with significant resolution improvements are anticipated. Introduction of the secondary emission conduction (SEC) stage to vidicons also will increase its sensitivity. Electromagnetic focusing and deflection rather than electrostatic focusing is planned for the improved vidicon, at the expense of weight and power for coils and the electronic drive circuitry. The most optimistic improvement forecasts a standard vidicon 50 mm x 50 mm target size with 80 lines/mm resolution (approximately 10,000-line tube) by 1975. The same tube, in a SEC version, would have less resolution, perhaps 50 lines/mm.

Basic research is being conducted on many concepts in imaging sensors or image storage devices. Typical areas include:

- Thermoplastic storage
- Photochromic dye storage
- Electrostatic tape recorders
- Vacuum-deposited silver halides.

There are no candidate imaging systems using these concepts that could be recommended at this time.

On the basis of the foregoing, it is concluded that any of the three candidate imaging systems cited could be available for the Voyager mission.

- 4) Impact on Spacecraft Reliability -- The three candidate imaging systems have comparable impact on the flight spacecraft with one exception. A tape recorder, or other means of data storage, is required for storing vidicon data. This tape recorder is not required for either the film or electrostatic tape cameras. The reliability of the tape recorder will lower the overall spacecraft reliability. Therefore, it is anticipated that the spacecraft reliability will be lower for the case where a vidicon system is used.

From contribution to mission success considerations, it appears that the film, tape, and vidicon systems are equally acceptable.

6.3.4.2 Performance of Mission Objectives

The key mission objective related to the Voyager imaging experiment is the attainment of the required Mars surface coverage to the specified resolution. A Voyager spacecraft system accommodating any one of the three candidate imaging systems can satisfy this objective. However, the flight spacecraft weights that must be allocated for achieving the key mission objective will differ for the three candidate systems. These weight differences result from both the imaging equipment itself and the demands imposed by the imaging equipment on other spacecraft hardware subsystems. The weight required by the imaging equipment and spacecraft hardware subsystems to achieve the mission objectives is a measure of the effectiveness of the system in performing the mission.

- 1) Impact of Resolution Requirements on Imaging System Weight -- The impact of ground resolution on the weight of film, vidicon (including data storage), and the electrostatic tape systems is shown in Figure 6-20. Ground resolutions are specified for a 500-km altitude.

All weights beyond the 300-pound region are considered very tenuous. The dashed lines for the vidicon and ESTC curves are based on resolutions as computed from the cutoff frequency for these sensors. The cutoff frequency represents the maximum theoretical resolution achievable. The nominal and conservative curves derate the operational frequencies from the quoted cutoff frequencies.

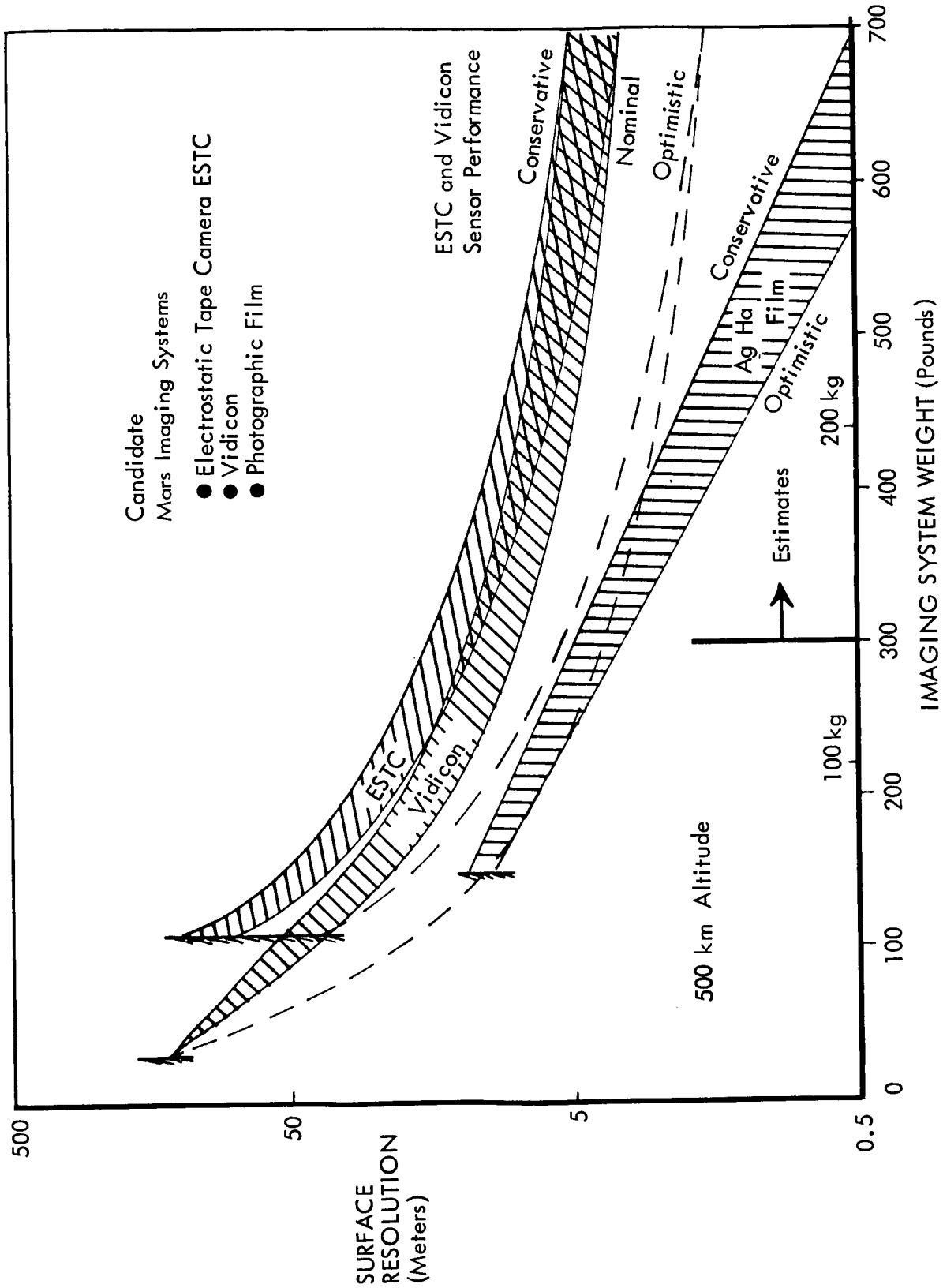


Figure 6-20: WEIGHT/RESOLUTION TREND FOR CANDIDATE IMAGING SYSTEMS

For the medium resolution requirements (100 to 300 meters), the vidicon system is the lightest, requiring approximately 30 pounds. For the high resolution requirements (1 to 10 meters), a film camera system requires the least weight. At 10 meters resolution, a film camera weighs 150 pounds versus 300 pounds for either the tape or total vidicon systems. The 1-meter resolution requirement will be met by a 500-pound film camera.

The vidicon and electrostatic tape camera probably cannot attain a ground resolution of 1 meter from a 500-km altitude. The highest achievable resolution for the vidicon and tape systems is predicted to be 5 meters from a 500-km altitude.

The other significant points of the weight versus resolution curve are the system base weights shown. These indicate the following:

- a) A suitable silver halide dual lens camera will not weigh less than 150 pounds (Lunar Orbiter weight) with 200 pounds being a more probable Martian camera system weight.
 - b) The electrostatic tape camera will weigh 110 pounds or more for a single camera without dual lens capability.
 - c) Very simple vidicon cameras will be possible at approximately 30 pounds (camera plus tape recorder).
- 2) Imaging System Impact on Spacecraft Subsystems -- The impact of an imaging system on the spacecraft hardware subsystems results from the following:
- a) Imaging system weight, which requires propulsion, structure, attitude control, and mechanism support.
 - b) Imaging system power, which requires solar array, battery, and thermal control support.
 - c) Imaging system data, which requires radio, telemetry, antenna, computing and sequencing, power, mechanism, data storage (vidicon only) support.

Imaging system weight and power are a direct function of the system resolution. Imaging system data generation depends on both system resolution and surface coverage. The impact of imaging resolution on imaging system weight was shown in Figure 6-20. Power requirements for the three candidate imaging systems for representative resolutions are summarized in Table 6-1. The electrostatic tape camera is seen to require the most overall power due to the high wattage requirements for long term readout.

The amount of information bits contained in each imaging frame is large. Current films contain approximately 10^9 bits per frame. Current vidicons contain only slightly more than 10^5 bits per frame. Electrostatic tape falls between the film and vidicon with approximately 10^7 bit per frame. Sensor data trends, in terms of bits per frame, are shown in Figure 6-21. The data indicate that vidicons will improve most in terms of information per frame by 1975. Even so, they will not be competitive with film systems.

IMAGING SYSTEM	PICTURE TAKING MODE Peak Power (watts)		READ OUT MODE Average Power (watts)	
	High Resolution	Medium Resolution	High Resolution	Medium Resolution
Vidicon	270***	16	45	15
Electrostatic Tape Camera	90	25	250 ***	30
Film Camera*	90**	60	90**	60

* Dual lens system

** Includes processing

*** Requires 200-watt electromagnetic
focus and deflection for
high resolution readout on
70 mm tape.

Table 6-1: TYPICAL POWER REQUIREMENTS FOR CANDIDATE IMAGING SYSTEMS

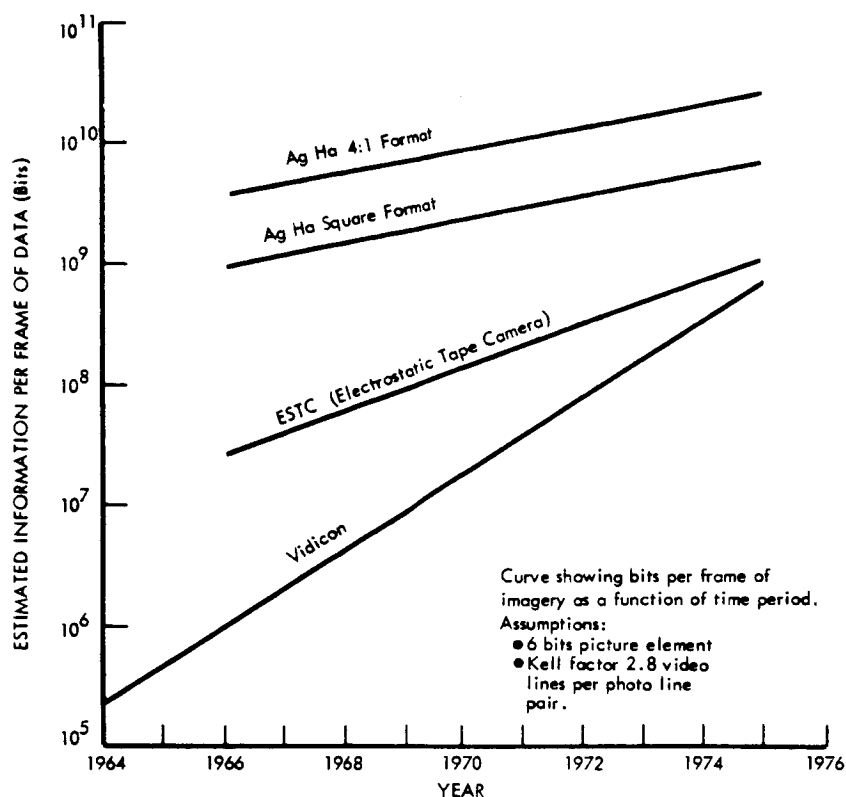


Figure 6-21: SENSOR DATA BIT TRENDS

The large amount of data generated by the imaging system will require high data transmission rates as can be inferred from the example given below for 100% coverage of Mars.

	Ground Resolution			
	1 meter	10 meters	100 meters	500 meters
Total Bits (54 bits/element)	7.66×10^{15}	7.66×10^{13}	7.66×10^{11}	3.03×10^{10}
Transmission time at 12,500 bps	24,400 yr	244 yr	29.2 mo	35.2 days

As shown, complete coverage of the surface of Mars at 1-meter resolution would generate more than 10^{15} bits of data. At a current state-of-the-art transmission rate of 12,500 bps, more than 24,000 years would be required to transmit the data to Earth. It is apparent that transmission of such high resolution coverage of Mars is impractical.

The 12,500-bps average data rate can be achieved for the 1973 Voyager mission. The required telecommunication subsystem would weigh over 300 pounds, and would include a 14-foot high-gain antenna and a 50-watt travelling wave tube, using digital modulation.

Anticipated state-of-the-art improvement would allow the attainment of a 50,000-bps data rate in 1975 for the same telecommunication subsystem weight.

By 1977, state-of-the-art improvements in conjunction with analog modulation techniques will provide data rates on the order of 300,000 bps at low weight penalties.

The above improvements in data rate capabilities will allow for increasing the photoimaging surface resolution and ground coverage percentage over those obtained in previous years.

The weight impact of increased photoimaging resolution on the spacecraft hardware subsystems is shown in Figure 6-22. The subsystems were divided into three categories: (1) provisions, which includes attitude control, data automation equipment, power switching, structure, thermal control and mechanisms, (2) power, and (3) propulsion. Telecommunications was not included as a separate category. As indicated above, data rate improvements for subsequent missions can be obtained with no incremental weight penalties. Increased telecommunication power requirements are reflected as weight increments in the power category.

The weight of provisions for attaining a photoimaging surface resolution of 100 meters is approximately 40 pounds (see Figure 6-22a). As this resolution is improved to 1 meter, the weight of provisions alone increases to 225 pounds. To obtain a given surface resolution, the lightest photoimaging unit was selected. Consequently, although not specifically indicated in the figure, the 100-meter resolution is obtained with a vidicon camera, whereas the 1-meter resolution is obtained with a film camera (see Figure 6-20).

The weight of provisions was found insensitive to the telecommunications data rate. This is not the case for the power subsystem, since the data rate increase is attained in part by employing more powerful amplifiers. A change from 100-meter resolution and a 12,500-bps data rate to 1-meter resolution and a 320,000 bps data rate increases the power subsystem weight attributed to imaging from 75 to approximately 180 pounds (see Figure 6-22b).

The additional orbital payload that must be accommodated by the propulsion subsystem to improve photoimaging resolution consists of improved photoimaging system, increased provisions, and increased power subsystem. This additional orbital payload is given in Figure 6-22c as a function of photoimaging resolution. As shown, a resolution of 1 meter requires an additional orbital payload of over 100 pounds for a spacecraft system with an average Earth-return data rate of 320,000 bps. The additional propulsion weight required to accommodate this additional orbital payload is shown in Figure 6-22d.

The total impact of increased photoimaging resolution and telemetry data rate on flight spacecraft is shown in Figure 6-23. If coverage of 1% of the Mars surface during a 180-day orbital mission is desired, then the highest resolution that can be accommodated by an average telemetry data rate of 320,000 bps is 4 meters. Also, as indicated in the figure, a film system is the lightest photoimaging unit for achieving that resolution. The overall weight impact on the flight spacecraft is approximately 1200 pounds. Figure 6-23 also shows the areas where the three candidate imaging systems offer the highest resolution per unit weight.

As indicated in Figure 6-23, coverage of a large percentage (in excess of 20%) of the planet at moderate to medium resolution (20 meters and higher) are best attained (from weight considerations) by vidicon systems. In this case, a problem exists

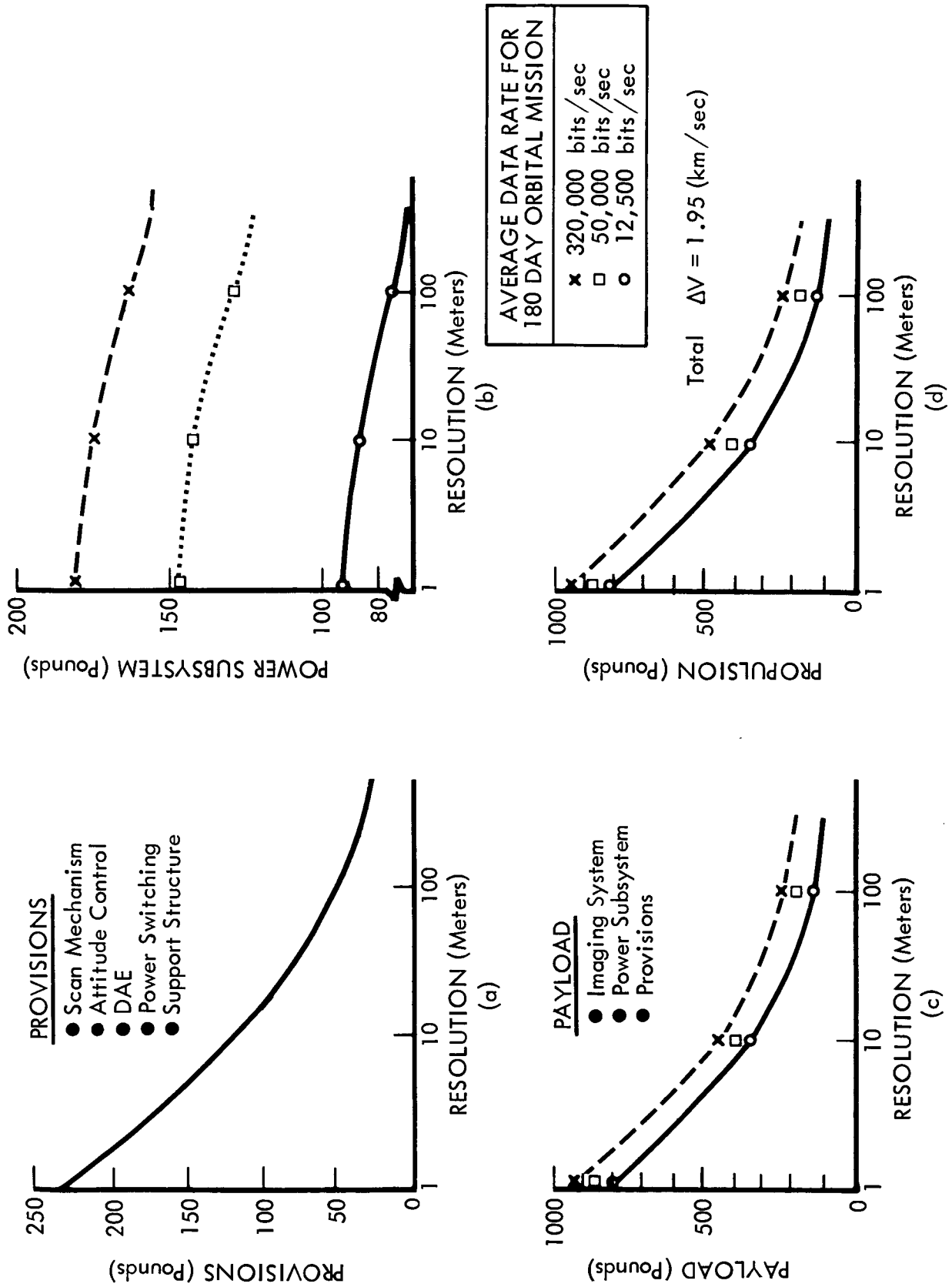


Figure 6-22: ESTIMATED IMPACT OF IMAGING SYSTEM RESOLUTION ON SPACECRAFT SUBSYSTEM WEIGHTS

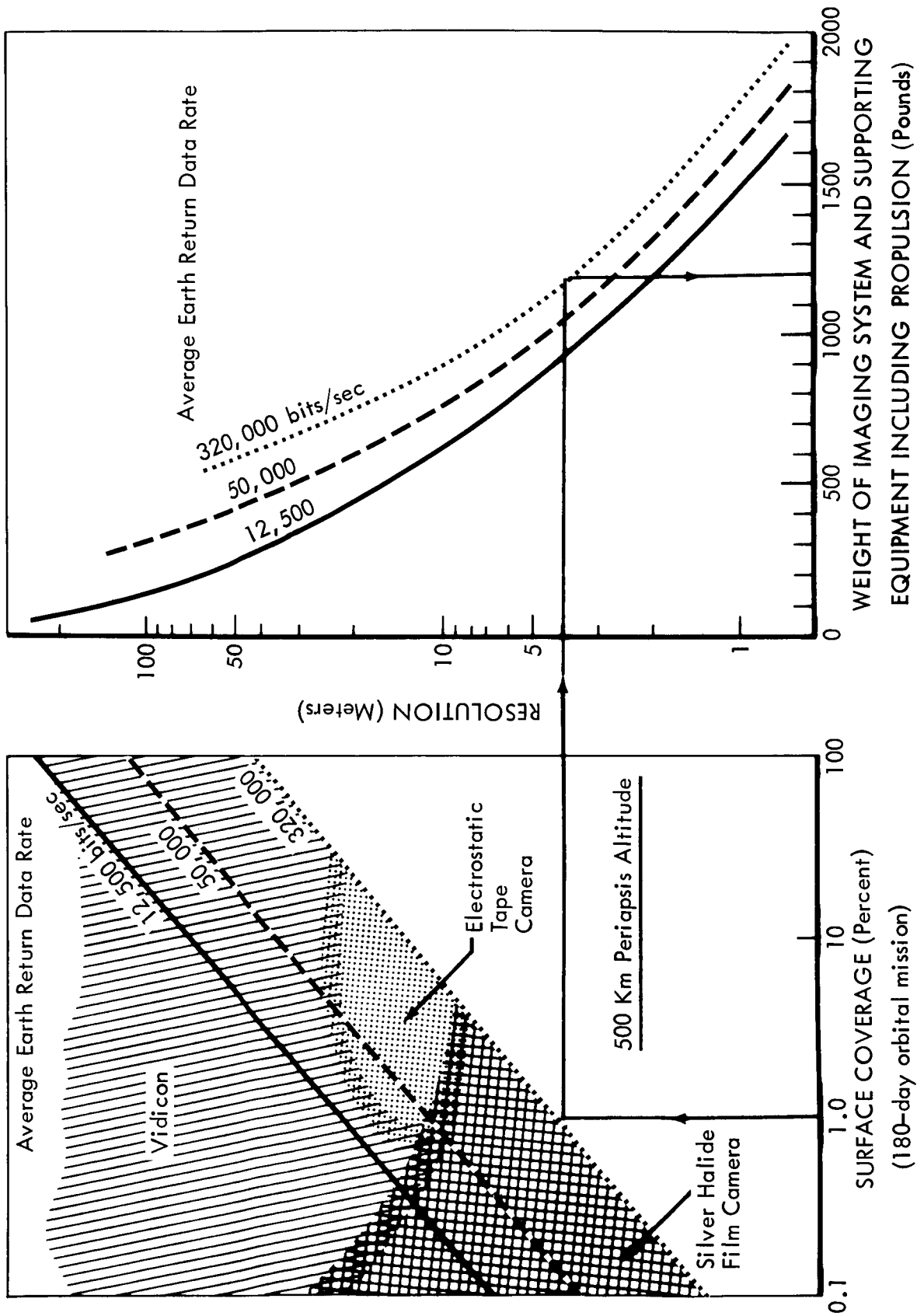


Figure 6-23: ESTIMATED IMPACT OF IMAGING SYSTEM HIGH RESOLUTION AND COVERAGE ON FLIGHT SPACECRAFT WEIGHT

in matching the vidicon data output with tape recorder input capability. A vidicon, particularly an advanced type, gathers bits onto the sensor surface at a very high rate during the picture exposure ($\sim 10^7$ bits per frame). The picture must then be scanned at a lower rate to read out these bits onto the tape recorder. The nominal 10-second scan time requires the readout at a faster rate than can be absorbed by the input read-in rate of available tape recorders. (It is anticipated that this scan time can be extended to 30 seconds without appreciable picture degradation.) The tape recorder(s) must be capable of storing at least one orbit's worth of photoimaging data bits, while the telemetry system must transmit to Earth, in one orbit, no less than one orbit's accumulation of data. Each of these interface "bottlenecks" must be carefully matched to ensure that (1) the system is not overdesigned, and (2) photoimaged data are not lost. This is illustrated in Figure 6-24. Significant tape recorder improvements are required. Otherwise, 2000- and 3000-line vidicons may not be feasible for 1973. A 1000-line vidicon would be feasible in 1973, provided tubes are developed for this application. The hypothetical science payload proposed for the 1973 Voyager Mars mission includes an imaging package consisting of three vidicon units. One of the vidicons is the high resolution camera. This camera has a slow cycling rate and does not aggravate the tape recorder problem cited above. The other two units are wide angle mapping vidicons mounted in fore-aft configuration to obtain convergent stereoscopic coverage. Such a configuration requires oblique imagery with 100% redundancy. Alternatively, as discussed earlier, spot coverage stereo may be obtained using orbit-to-orbit roll of the spacecraft. Repeated coverage for determining seasonal changes also will provide stereo data. A different use of the two mapping cameras, in the hypothetical 1973 photoimaging payload therefore is proposed to alleviate the video data impact on the data storage subsystem. It is proposed to mount the two mapping vidicon cameras with their optical axes parallel and vertical, i.e., along the local planetary vertical during imaging. The cameras are used in an alternate exposure mode to obtain a contiguous coverage swath.

As previously stated, the 3000-line vidicon severely taxes the tape recorder when a 10-second readout time is imposed. However, when used in an alternate exposure mode, and allowing as much time for vidicon readout as permitted by orbital geometry, the cycling and data rates will be as given in Table 6-2.

The required data rates of 818,000 bps for 100-meter mapping, and 409,000 bps for 200-meter mapping still exceed the 100,000 and 200,000 bps rates quoted by tape recorder manufacturers. However, a limited tape recorder breakthrough in the problem area of track-to-track data skew for high density recorders (10^4 bits/inch/track) may occur. This could allow parallel track recording at rates compatible with the vidicon output rates cited.

An additional problem is associated with this proposed alternate. It required 70 to 140-second slow scan readout from the vidicon. These long scan times have been checked with vidicon manufacturers. The consensus is that such long scan times are probably within the state of the art if some resolution loss is accepted.

Weight, however, can compound this problem. The vidicons cited for the 1973 hypothetical payload are assigned weights that suggest electrostatic focus and deflection control. Discussions with tube manufacturers imply that vidicons and SEC vidicons having the resolutions cited may not be available in an electrostatic version until perhaps 1975. Should the electrostatically focused vidicons not be available for the 1973 mission, then heavier electromagnetically focused vidicons would be used. In this case, either (1) the number of wide angle vidicons must be decreased from two to one, or (2) the weight allocation for the science payload must be increased.

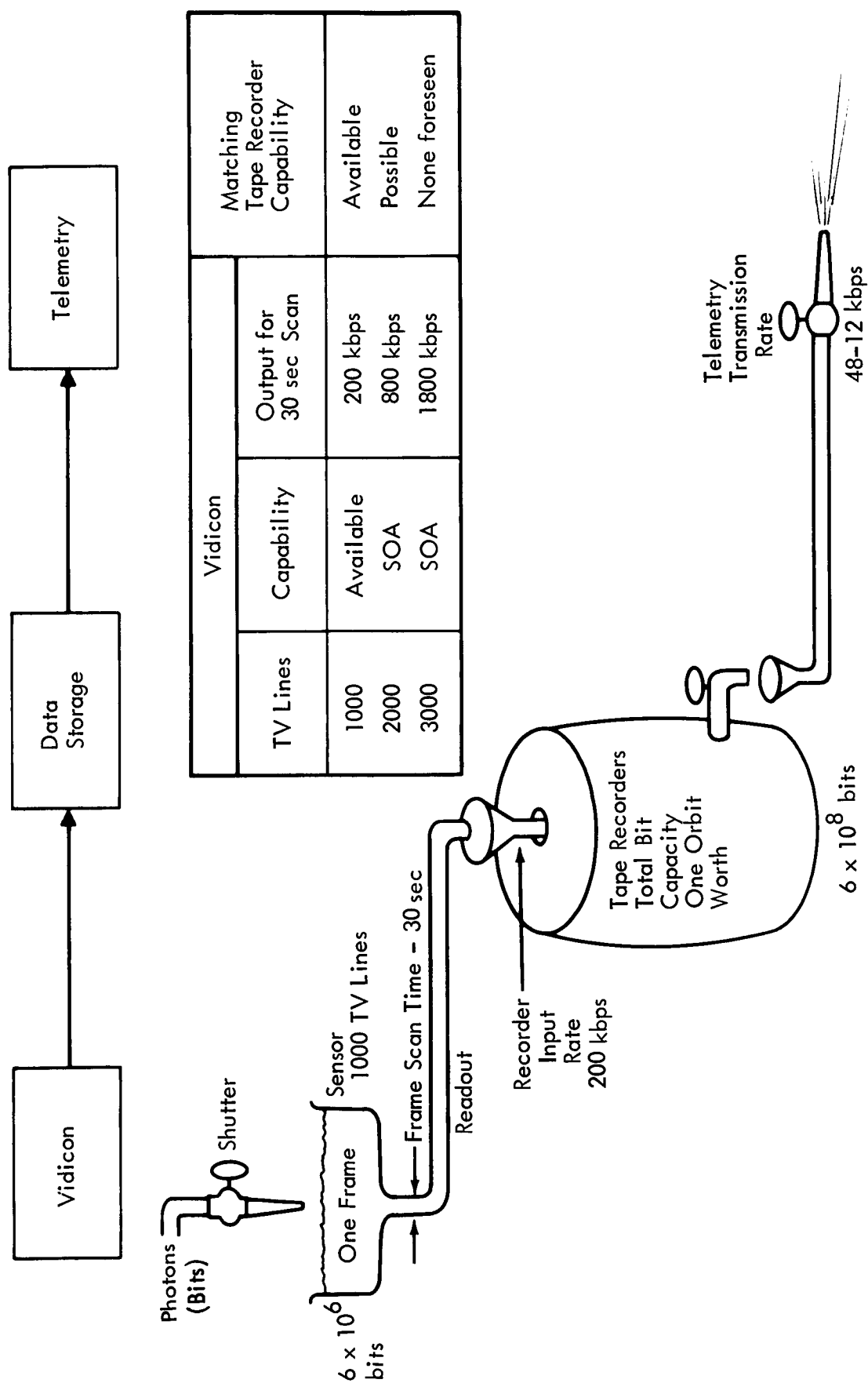


Figure 6-24: VIDICON / TAPE RECORDER COMPATIBILITY

Table 6.2: ALTERNATE VIDICON EXPOSURE MODE

Vehicle Altitude (km)	Focal Length for 100-Meter Mapping	Size of Area Mapped per Frame	Cycling & Data Rate	Focal Length for 200-Meter Mapping	Size of Area Mapped per Frame	Cycling & Data Rate
1500	27 cm	140 by 140 km	69.6 sec (66 sec with 5% frame overlap)	13.5 cm	280 by 280 km	140 sec (132 sec with 5% frame overlap)
1000	18 cm			9 cm		
500	9 cm		Readout rate = 818,000 bits/sec	4.5 cm		Readout rate = 409,000 bits/sec
300	5.4 cm			2.7 cm		

One inch - 3000 line Vidicon with 54×10^6 bits per frame.

6.4 CONCLUSIONS

The following key conclusions have been reached as a result of the photoimaging consideration study.

- 1) High resolution (1 to 10 meters) photoimaging coverage of a small fraction of Mars' surface (0.1% to 1%) and medium resolution (150 to 300 meters) photoimaging coverage of most of the planet's surface will satisfy the presently understood scientific objectives.
- 2) The film camera, vidicon, and electrostatic tape camera systems can satisfy the nominal photoimaging resolution and coverage requirements.
- 3) The film camera system provides the highest resolution of the three candidate imaging systems.
- 4) For the lowest allowable orbital altitude of 500 km (from planetary quarantine considerations), the highest resolution of a film camera is estimated as 0.5 meter, limited only by the postulated scattering, turbulence, and aerosol phenomena of the Martian atmosphere and not by equipment capability. This 0.5-meter resolution is achievable with a film camera weighing less than 700 pounds.
- 5) The flight spacecraft weight **impact associated with the 700-pound film camera** is 1800 pounds. This would allow for approximately 0.01% coverage of the planet at the 0.5 meter resolution over a 180-day orbital mission.

- 6) For the lowest allowable orbital altitude of 500 km, the highest achievable resolution of either the vidicon or electrostatic tape system is estimated to be 5 meters due to sensor/lens limitations.
- 7) For the lowest allowable orbital altitude of 500 km, the film camera system will have the least impact on spacecraft weight for resolutions below 10 meters.
- 8) The vidicon imaging system will satisfy the medium resolution requirements for the least weight.
- 9) For high resolution imagery, modified Cassegrainian-type reflective telescopes using folded optics will provide the required effective focal length of 6 to 12 meters within the spacecraft envelope constraint.
- 10) Film processes have been developed with a demonstrated life capability in excess of 2 years under simulated space conditions.
- 11) The SEC vidicon, because of its high sensitivity, will result in the lightest, most compact optical system by comparison with other sensors.
- 12) Advanced 3000-line slow-scan vidicons (~140 seconds) and high read-in rate tape recorders (~400,000 bps) should be developed to satisfy the requirements of the hypothetical photoimaging payload currently proposed for the 1973 mission.

ERRATA SHEET

<u>Page</u>	<u>Errata</u>
2-19	Table 2-3 in the second line of the note below the table, change "thest" to "these".
2-21	In the second paragraph, fourteenth line, change "mHz" to "MHz".
3-2	In figure 3-1, add "task No." above each of the blocks in the flow diagram and in each of the numbers delete the ".0".
4-44	In the second paragraph, second line, change "mission" to "missions".
5-3	In item 3 of the fourth paragraph, change "sp cecraft" to "spacecraft".
6-6	In the first paragraph, fourth line, change "transmittal" to "transmitted".